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A CONCEPTUAL DESIGN OF THE SPACE SHUTTLE INTEGRATED AVIONICS SYSTEM

By Electronics and Controls Division
Edited by Fred E. Digesu

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This report presents the status of an investigation to define an integrated avionics system for the Space Shuttle. The system concept depicted for the integrated approach, based on subsystem requirements established early in the study, meets the mission requirements outlined in Shuttle Task Group reports and is compatible with the significant features of the vehicle configurations resulting from several 1969 contractor design studies. Included as part of the integrated avionics system is all Shuttle equipment that utilizes electronics in performance of the various functions required to accomplish the Space Shuttle mission. The individual subsystems are defined in varying degrees of detail and all data will be subject to change as the design work continues. Sufficient data are presented, however, to identify the significant physical characteristics, important performance values, functional block diagrams, and preliminary hardware selection for each avionics subsystem.

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SUMMARY

This report presents the status of an investigation to define an integrated avionics system for the Space Shuttle. The system concept depicted for the integrated approach, based on subsystem requirements established early in the study, meets the mission requirements outlined in Shuttle Task Group reports and is compatible with the significant features of the vehicle configurations resulting from several 1969 contractor design studies. Included as part of the integrated avionics system is all Shuttle equipment that utilizes electronics in performance of the various functions required to accomplish the Space Shuttle mission. The individual subsystems are defined in varying degrees of detail and all data will be subject to change as the design work continues. Sufficient data are presented, however, to identify the significant physical characteristics, important performance values, functional block diagrams, and preliminary hardware selection for each avionics subsystem.

I. INTRODUCTION

The shuttle task group report, dated June 1969, described a number of desirable features that should be included in a Space Shuttle. Since the publication of that report, a great deal of effort has been expended in NASA and in industry toward further definition of both requirements for a shuttle and of systems concepts to meet those requirements.

This report documents the status of an approach to developing the Integrated Avionics System requirements and a means for meeting them.

Definition Of Terms

The Shuttle Avionics System has been referred to by the various groups working the problem as

- Shuttle Avionics System.
- Integrated Electronics System.
- Integrated Avionics System.
- Shuttle Astrionics System.

The term Integrated Avionics System (IAS) will be used in this report.

Included as part of this system is all shuttle equipment that uses electronics to sense, acquire, generate, transmit, process, store, record, and display data required in the operation of shuttle systems, for determination of operational and flight-ready status, for performing onboard launch and mission control function, and for assisting in or verifying maintenance activity. Also included is any equipment associated with production, distribution, storage, and regulation of electrical power. The software required to accomplish the functions described above is considered an integral part of the IAS.

Work Scope

Preliminary investigations by both NASA and industry indicate that a completely integrated avionics system should

- Improve system reliability.
- Reduce turn around time between flights.
- Minimize crew size.
- Reduce overall program costs.

The proper way to quantize the benefits resulting from development of a completely integrated system would be a detailed design study of both conventional and integrated approaches to the system, and a comparison through tradeoff of the approaches within overall program tradeoffs. The accomplishment of this rather formidable task requires an establishment of system functional and performance requirements, a design meeting these requirements for each approach, and finally overall comparison of these.

This report develops a system concept for the integrated approach based on a set of subsystems requirements established during the early phases of the work. Note that there are no firm requirements on shuttle performance, configuration, or on the avionics system at present. For this reason, many assumptions were made to permit the investigation to proceed.

Further, note that this report does not cover the conventional approach and the comparison between conventional and integrated approaches. It is hoped that future work will complete the picture, using the same set of assumptions used here.

Background

Lockheed Missiles and Space Company (LMSC), General Dynamics Corporation (GDC), Martin Marietta Corporation (MMC), and other contractors carried out separate tasks under which they were to investigate certain aspects of the Integrated Avionics System for a representative vehicle. Concurrent in-house investigations were carried out while supervising these tasks. Material generated by these three activities, as well as the concepts from other NASA and industry groups, are incorporated in this system conceptual design report.

Subsystem Identification

One of the principal questions concerning the IAS is how the interfaces between and within the various shuttle systems are to be managed. A top-level system description was developed to establish an overall framework within which the different functional and performance requirements could be identified and related. Table 1-1 gives the shuttle systems breakdown used.

TABLE 1-1. SYSTEMS BREAKDOWN

Shuttle System	Subsystem Number
Structural/Mechanical	1.0
Propulsion	2.0
Electrical Power	3.0
Environmental Control	4.0
Guidance, Navigation, and Control	5.0
Communication	6.0
Controls and Displays	7.0
Data Management	8.0

This breakdown, together with its lower levels (e.g., first level is 1.1), proved extremely valuable. It is recommended that this structure be retained in all future work.

II. IAS GENERAL REQUIREMENTS

Mission Requirements

The nominal mission assumed was the space station logistics resupply mission. This mission is characterized by the following:

Self-sustaining lifetime — 7 days

Stay time at space station — 5 days

Vehicle lifetime — 100 missions

Launch window — 60 seconds

The mission profile used for this report is shown in Figure 2-1. The vehicle configuration does not drive the IAS design significantly.

The nominal mission also assumed various phases for both booster and orbiter. The flight phases for the booster are (1) lift-off, (2) atmospheric boost, (3) separation, (4) coast, (5) reentry, (6) glide or cruise, and (7) landing. The flight phases for the orbiter are (1) separation, (2) guided burn, (3) orbit, (4) reentry, (5) glide or cruise, (6) landing, and (7) self-ferry.

Vehicle Configuration

The vehicle configurations developed by three of the contractors involved in space shuttle work are shown in Figures 2-2 through 2-10. All are two-stage vehicles, using multiple rocket engines in each stage. The GDC vehicle is the only one which employs variable geometry, in addition to multiple jet engines which are used to extend descent range and to improve approach and landing characteristics. Conventional wheel landing gear is used throughout.

Mass, geometric, and aerodynamic characteristics of these vehicles, which are necessary for IAS design studies, are shown in Appendix B.

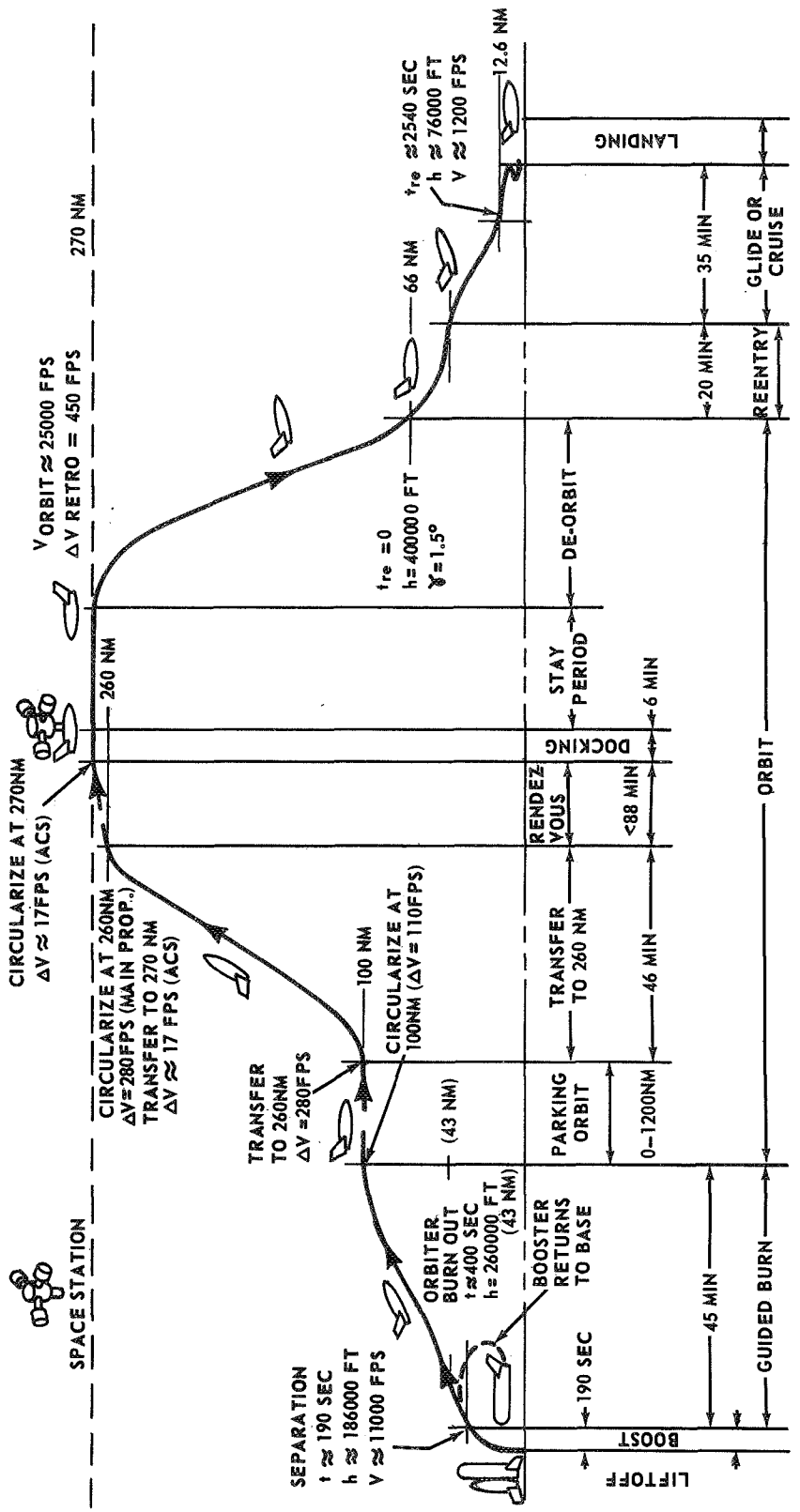


Figure 2-1. Nominal profile mission.

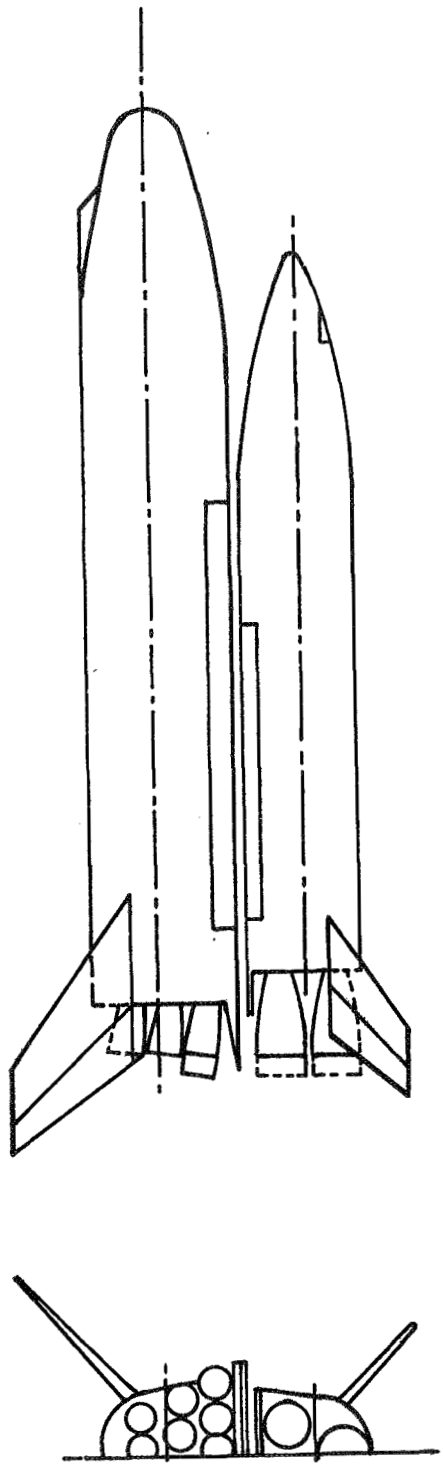


Figure 2-2. GD/C FR-3 launch configuration.

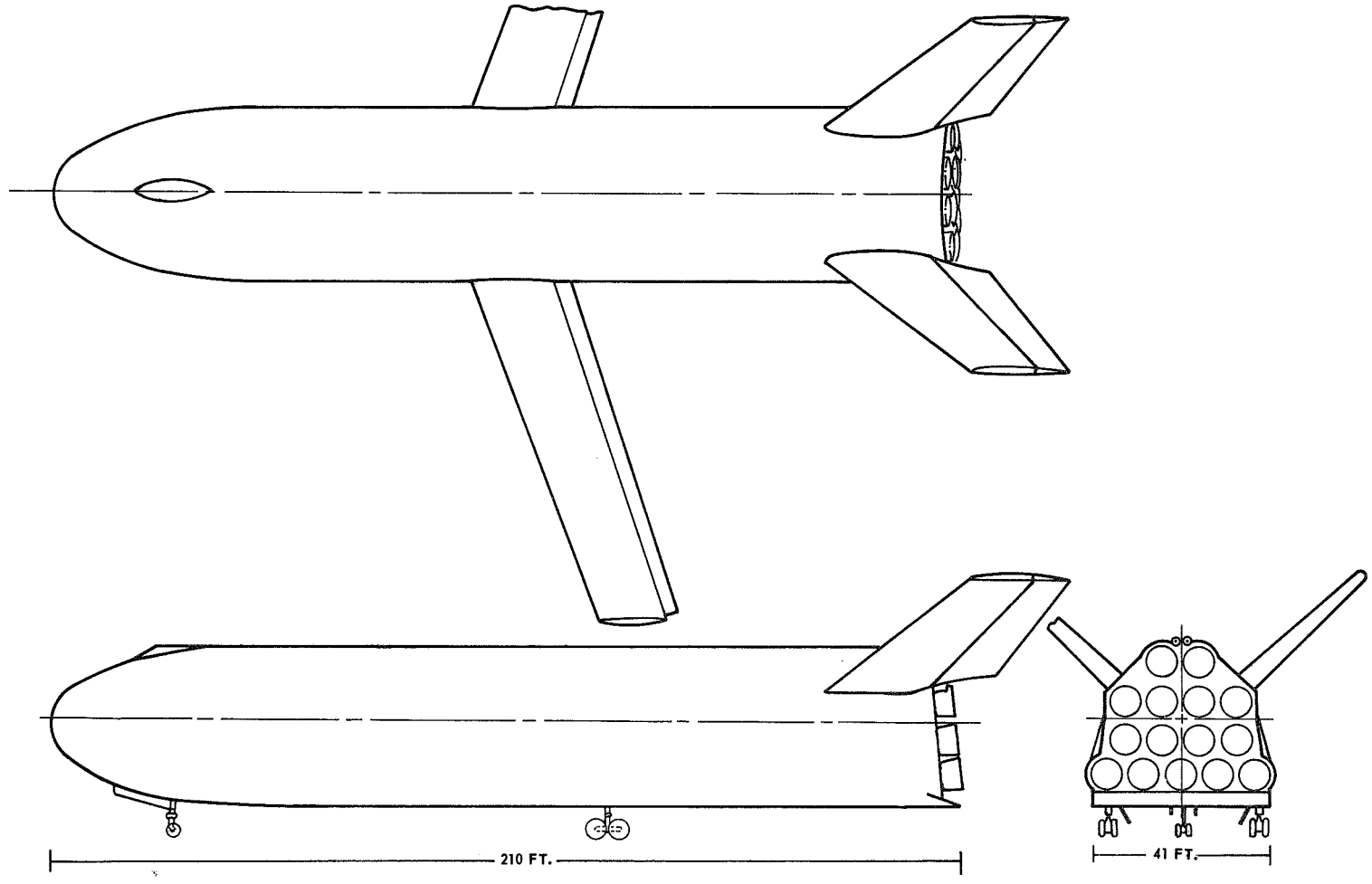


Figure 2-3. GD/C FR-3 booster basic configuration.

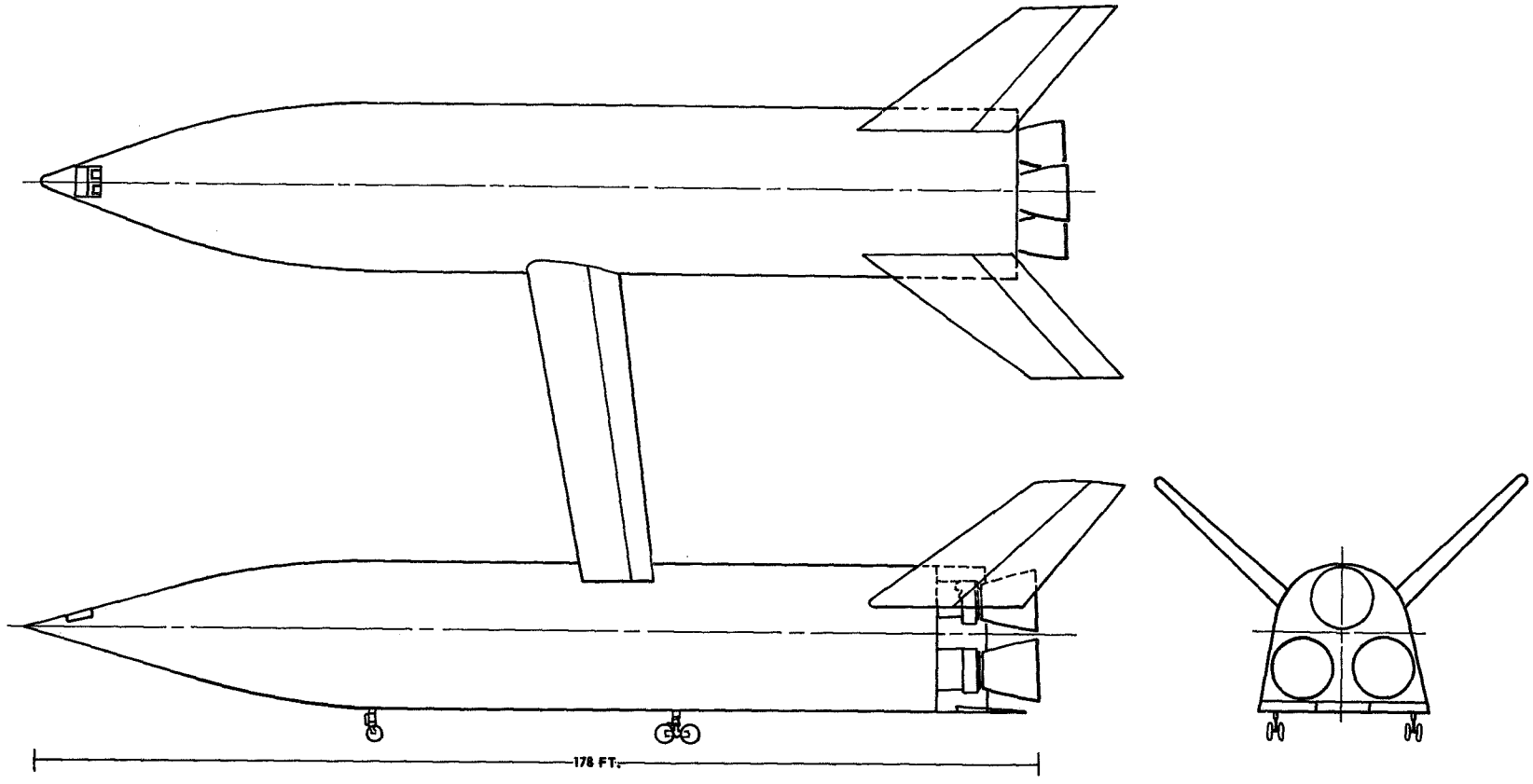


Figure 2-4. GD/C T-18 orbiter element.

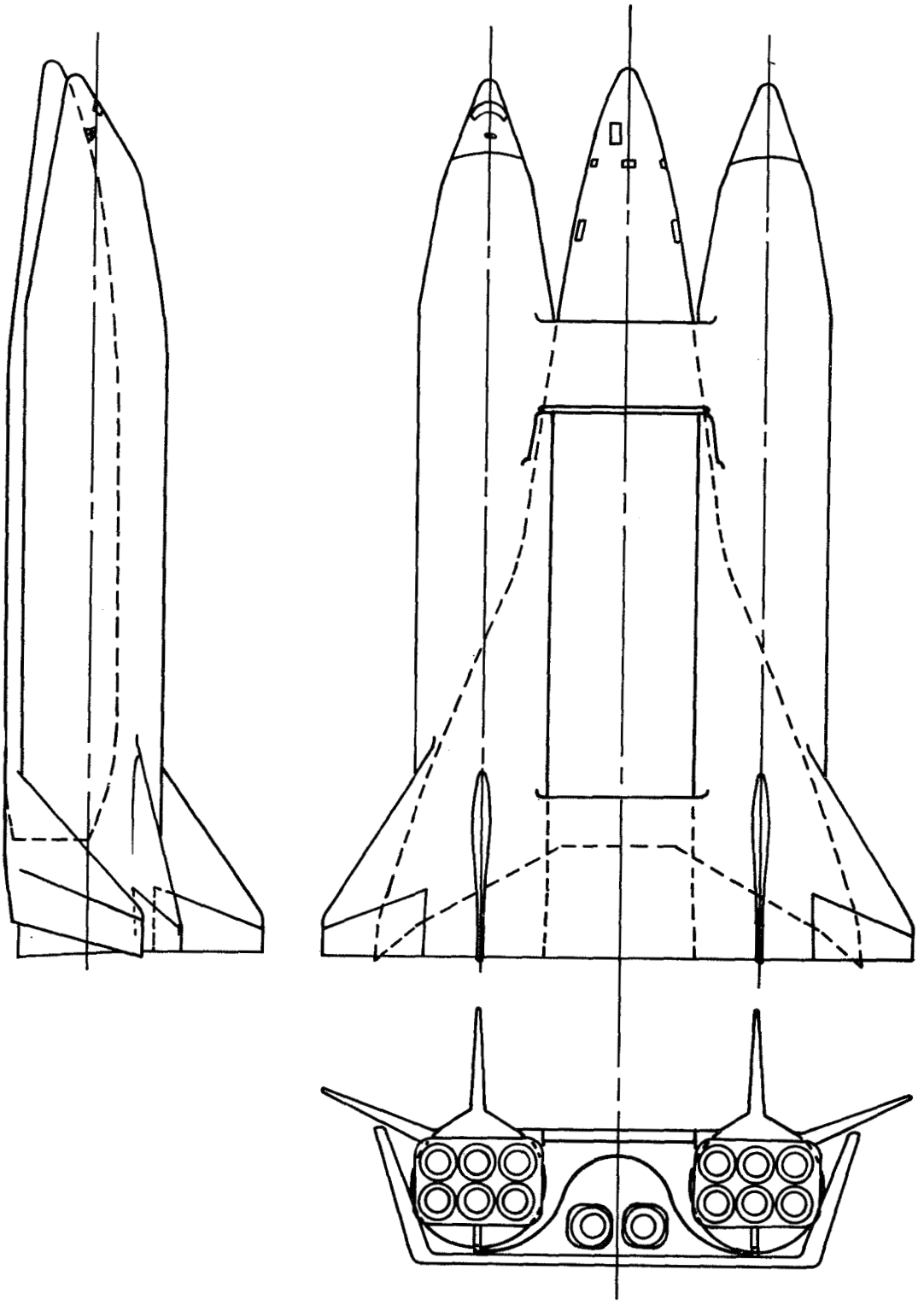


Figure 2-5. Martin Marietta two-stage launch configuration.

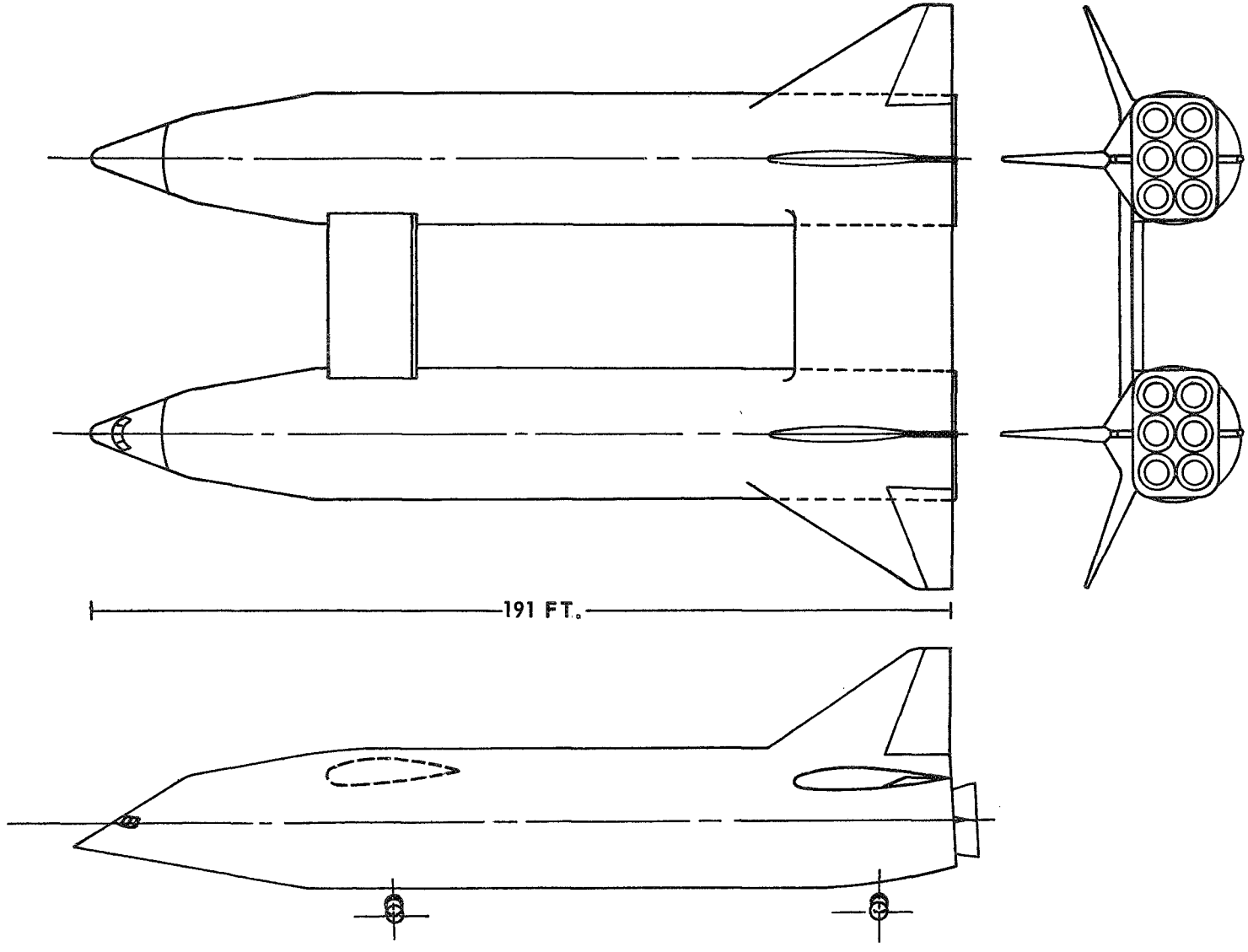


Figure 2-6. Martin Marietta two-stage spacecraft, Stage I.

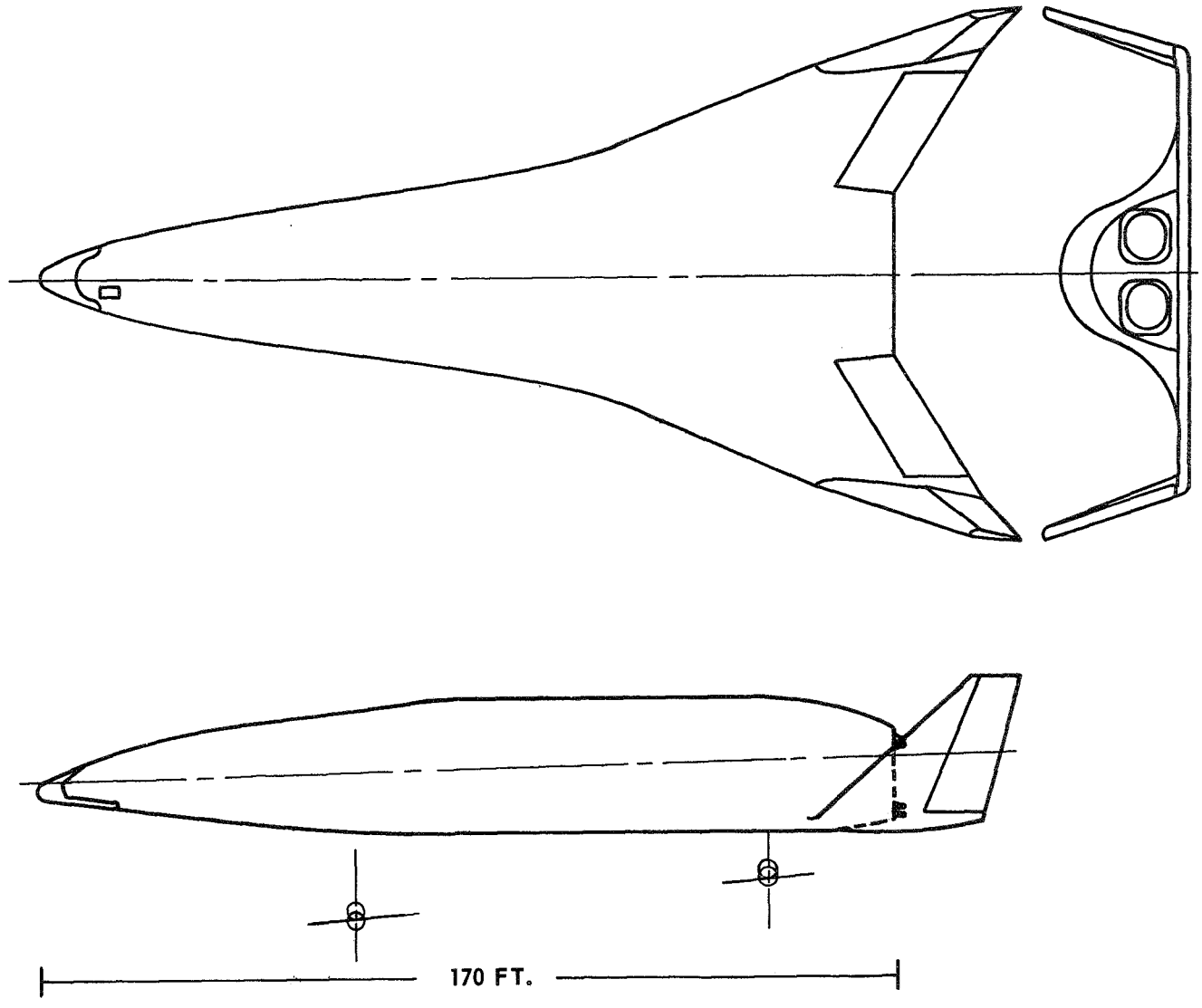


Figure 2-7. Martin Marietta two-stage spacecraft, Stage II.

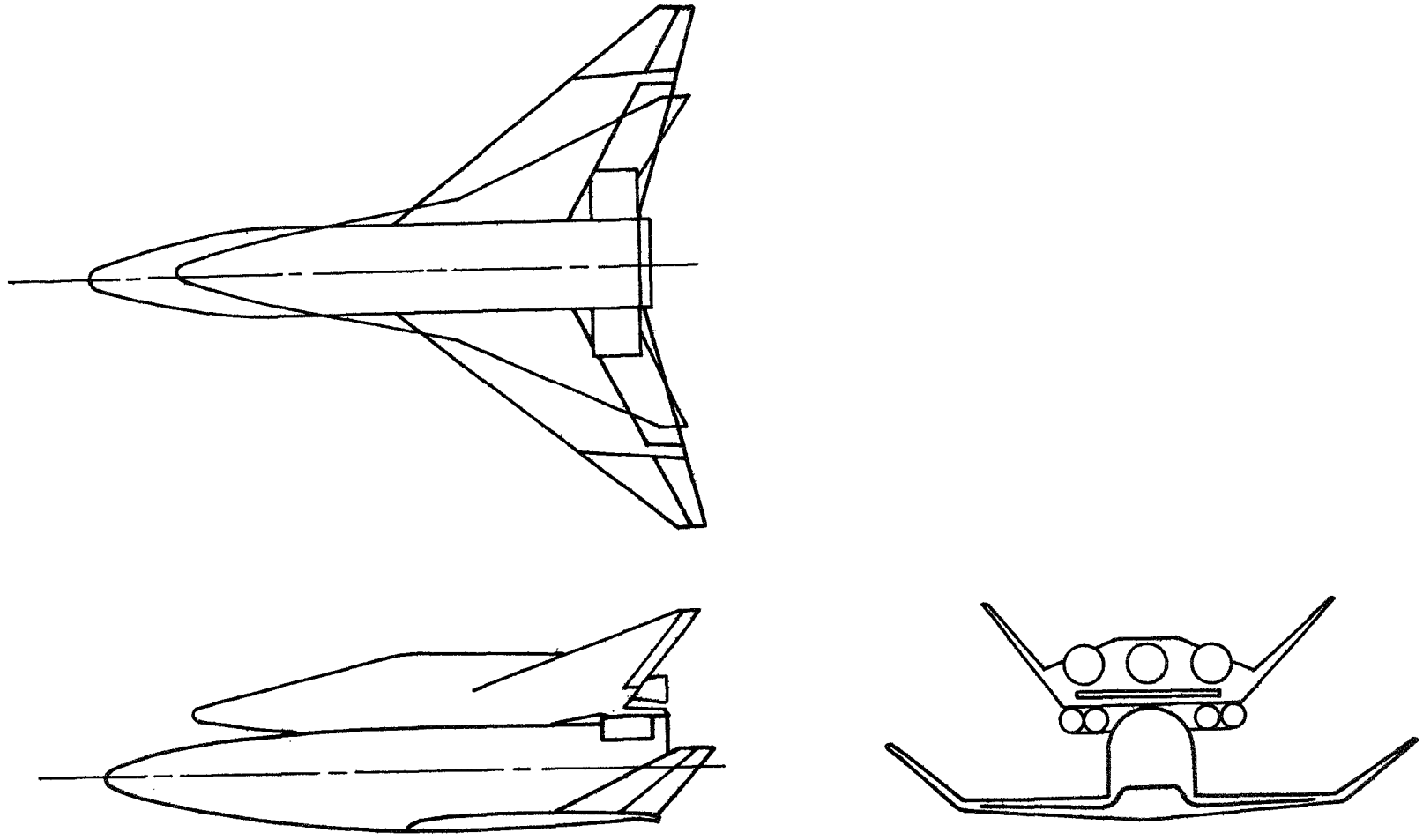


Figure 2-8. Composite LMSC launch vehicle general arrangement.

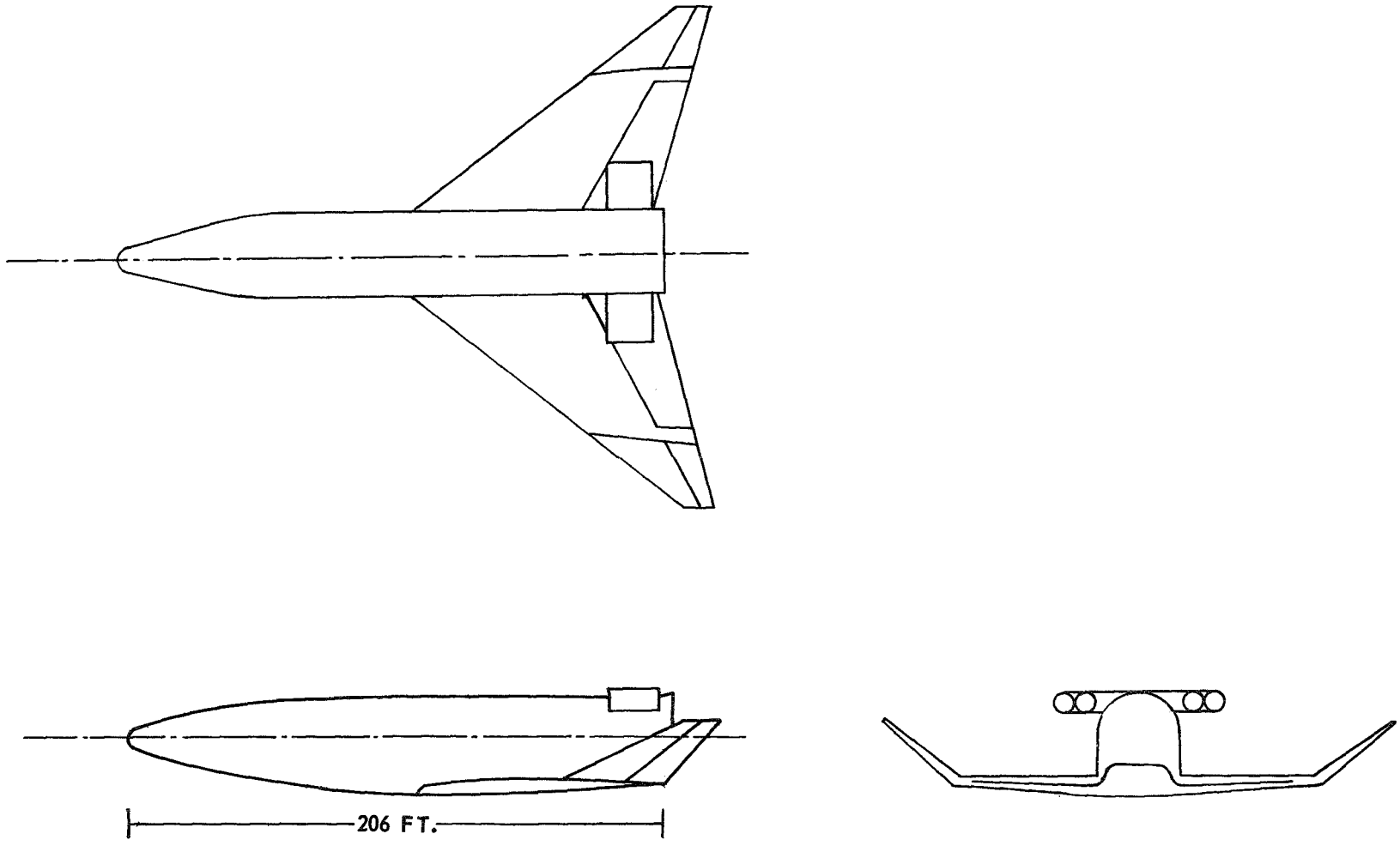


Figure 2-9. Lockheed return booster general arrangement.

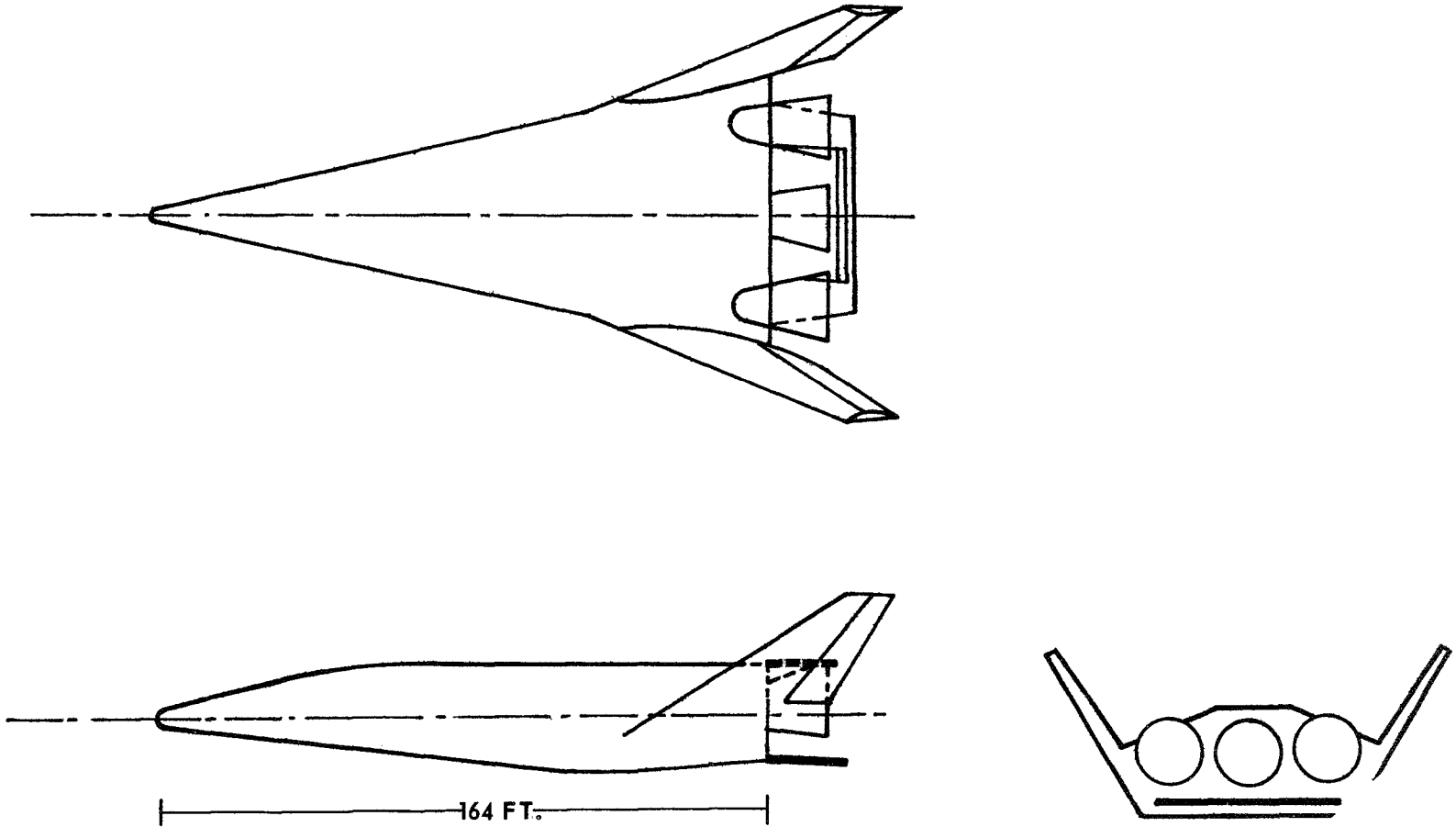


Figure 2 10. Lockheed spacecraft general arrangement.

Autonomy

The shuttle is required to be capable of autonomous operation through all mission phases and operating modes. There will not be any large-scale ground support such as mission control and ground tracking net operation. This means, for example, in-orbit navigation capability, no high-rate telemetry to the ground, minimum GSE during launch, and maximum onboard checkout and fault isolation, plus abort situation warning capability must all become features of the IAS. For the post-reentry and landing phases for both booster and orbiter, however, total ground independence is not possible since extensive use must be made of existing civilian and military radio navigation and landing aids.

Reliability

Reliability is of paramount importance in the design of the Integrated Avionics System (IAS) if the requirements of safety, long operating life, and total number of shuttle missions are to be met. The Space Shuttle Task Group (STG) report imposed a reliability requirement of "fail operational, fail operational, fail safe" on the space shuttle IAS. In strictest terms, if this requirement is to be implemented at the line replaceable unit (LRU) level, it means quadruple redundancy.

Normally, the degree of redundancy is dictated by determining the reliability or criticality number required for a subsystem or the performance of a function in view of a reliability goal for the overall system. This reliability is then achieved either by redundancy or by designing the subsystem to be sufficiently reliable. If redundancy is utilized, the required failure detection as well as switching techniques should not significantly degrade the gain in reliability achieved by using redundant hardware; and the failure detection logic and safing devices should not have failure modes which could terminate a satisfactory subsystem to the detriment of the mission. The best approach is to design the simplest system which meets the required overall reliability goal. It is, therefore, recommended that the ground rule of "fail operational, fail operational, fail safe" be replaced with a reliability goal for the IAS, to be met in the best way dictated by detailed analysis and design.

Maintenance

All subsystems should be designed for minimum maintenance, modular replacement, and maximum usage of standard aircraft type maintenance. The shuttle vehicle design should permit rapid maintenance through simplified replacement of parts and systems where feasible; removal and replacement time should be minimized with onboard checkout and module accessibility.

Every effort must be made to provide maximum reusability, to minimize expendables, and to provide minimum turnaround time. Turnaround time after a nominal mission (from landing to launch readiness) should be restricted to two 40-hour work weeks. If necessary, this turnaround time can be reduced further by employing multiple work shifts to condense the 80-hour work period into 5 calendar days.

III. IAS DESCRIPTION

Since no particular shuttle system configuration can be proven better than another at this time, features of each were used to assemble the IAS requirements. The space station logistics resupply mission was used as a basis for the system description undertaken here. Two major aspects have been attacked: (1) description of the various subsystem concepts; and (2) description of the integration and interface management concept.

The major decision to be made in design of an IAS is the interface management between and within subsystems. For this reason, the approach taken has been to identify, on a subsystem by subsystem basis (reference Subsystems 1.0 through 8.0 of the section), the data and control signal flow both within and between subsystems. Once sufficient understanding of these requirements is attained, it becomes possible to work on the integration aspects of the IAS. This section covers the definition and description of each subsystem, including the overall integration and data distribution portion of the problem. Since Subsystems 1.0, 2.0, and 4.0 are not avionics subsystems, no details on these subsystems are provided in this report.

1.0 Structural/Mechanical

Considerable information and data on the structural/mechanical subsystem of the space shuttle vehicle have been generated by various contractor studies [1, 2, and 3] as well as by in-house NASA investigations during the Phase-A Integral Launch and Reentry Vehicle (ILRV) studies. This available reference material was freely used to determine the necessary background and interface information to permit the IAS design to be pursued. As an example, the structural/mechanical subsystems as generated by LMSC are illustrated by Appendix D, 1.0. Primarily, requirements, as well as constraints, imposed on the IAS by the structural/mechanical subsystem of the various configurations were identified and utilized where necessary in the respective IAS subsystem studies and investigations.

2.0 Propulsion

Considerable information and data on the propulsion subsystem of the space shuttle vehicle have been generated by various contractor studies [1, 2, 3] as well as by in-house NASA investigations during the Phase-A ILRV studies. This available reference material was freely used to determine the necessary background and interface information to permit the IAS design to be pursued. As an example, the propulsion subsystems as generated by LMSC are illustrated by Appendix D, 2.0. Primarily, requirements, as well as constraints, imposed on the IAS by the propulsion subsystem of the various configurations were identified and utilized where necessary in the respective IAS subsystem studies and investigations. Although many different engine configurations for the various vehicle concepts have been proposed, an engine configuration using 400K-pound thrust engines, with 12 engines on the booster and 3 engines on the orbiter was selected to maintain consistency throughout the various subsystem analyses.

Additional information has been developed by Pratt and Whitney and the Bendix Corporation on engine control and monitoring functions [4]. In future work, the complexity imposed on engine control by the requirement to precisely control thrust should trade the IAS capability to carry out this function against an engine-mounted control system. Further trades should be made to determine if precise throttling is really required.

3.0 Electrical Power System (EPS)

3.1 General Vehicle Requirements

The shuttle vehicle must be supplied with electrical power continuously from lift-off through the landing phases. The orbiter has the more difficult power requirement because of the 7-day mission period. The type of mission and the vehicle requirements appear to favor electrochemical power sources rather than the electromechanical types.

After reentry, alternators driven by the turbojet engines will supply the major electrical power for the aircraft mode of the mission. If the vehicle selected does not have landing engines, the electrical power required will be supplied from batteries and/or fuel cells.

3.1.1 Design Requirements

The electrical power system shall provide full operational capability, without transients or degradation of the power quality, after the first failure. After the second failure, the electrical power system must provide sufficient power to maintain a safe condition. All equipment required for a safe return must be powered. These redundancy requirements apply to all elements of the electrical power system: generation, conversion, and distribution.

3.1.2 Design Objectives

The operating life of the vehicle is 10 years. Equipment used should be able to operate reliably during the use period and have minimum degradation for stored time between flights. To achieve this goal, a minimum number of major components should be used in assembling the system, each possessing high reliability. The reusable aspect of the vehicle requires that selected major components need only minimum scheduled maintenance to reduce the annual cost of the vehicle. Unscheduled maintenance will be consistent with reliability achieved in operation.

3.2 Power Requirements

The shuttle peak power requirements are summarized in Table 3-1, as derived from analysis by each responsible vehicle subsystem area. The

TABLE 3-1. SPACE SHUTTLE ELECTRICAL PEAK LOAD SUMMARY

COMM & DATA MGMT	WATTS
COMPUTER	350
DATA BUS INTERFACE	300
COMMUNICATIONS (VOICE, TELE. DATA & NAVIGATION AIDS	2800
	3450

ENV. CONT. & LIFE SUP.	WATTS
WASTE MANAGEMENT	150
THERMAL CONTROL	800
ATMOSPHERE CONTROL	515
HYGIENE	170
WARNING & CONTROL SYSTEMS	35
SUIT BLOWER	200
	1870

ELECTRICAL SYSTEM	WATTS
ELECTRICAL DISTRIBUTION	680
CONTROLS AND DISPLAYS	2000
	2680

GUIDANCE NAV. & CONTROLS	WATTS
SENSORS & SIG. COND.	425
CONTROL ELECTRONICS	300
	725

PROPULSION ENGINE XLR-129 (1 ENG.)	WATTS
CONTROL POWER	5000
EXPANSION NOZZLE	900
INSTRUMENTATION	40
	5940

control power requirements for the XLR-129-1 propulsion engine were furnished by the contractor [4]. Power requirements for environmental control and life support were furnished by PD-DO-ME [5]. Conferences were held with each responsible subsystem area to determine which units would be operated in the various phases of the mission. Certain systems, such as lighting, were considered to operate continuously for all phases. The average power required for guidance, navigation, and control; communications; and environmental control and life support were reduced from peak values by a factor of 0.75. The average powers for multiplexing and engine control were reduced from peak power values by a factor of 0.2. These average amounts of power have been translated to power profiles for each specific phase of the mission in Figures 3-1 and 3-2. The profiles for prelaunch and post-landing checkout periods reflect all equipment on, and control power for one engine. The control power for the propulsion engine is battery supplied, for one second, during initial start. Once started, engine control power is supplied by an engine driven power source. No electrical power is included in these profiles for aerodynamic surface control, deployment of landing engines, and landing gear. These short-term high-peak hydraulic power requirements will be supplied by a chemically-fueled turbine-driven system.

Based on an analysis of the loads and the length of the mission, the electrical power requirements will be about equally divided between ac and dc for the booster. The electrical loads for the orbiter will be about 25 percent alternating current and 75 percent direct current. Total power requirements were adjusted upward 5 percent to account for distribution losses. The overall efficiency for the inverters used was 85 percent. The alternating current loads are given in Table 3-2.

TABLE 3-2. ALTERNATING CURRENT LOADS

Description	Volt-Amp
Fuel boost pumps	9 000
Air data, turn and bank	400
Fuel flow, fire detectors	300
Ice detectors, anti-ice scoop	5 300
Fuel and oil, quantity, pressure ratio	200
Anticollision lights	500
Landing lights	4000
Transformer/rectifier ac load	3 500
	23 000

OPERATE TIME: 3.1 HRS
AVG POWER: 6.0 KW
TOTAL ENERGY: 18.6 KWH
PEAK POWER: 26.0 KW

- PRIMARY-BATTERY SYSTEM SILVER-ZINC
- TURBO SYSTEM FOR PEAKS
 560 HORSEPOWER TO HYDRAULICS
 FOR 6 MINUTES

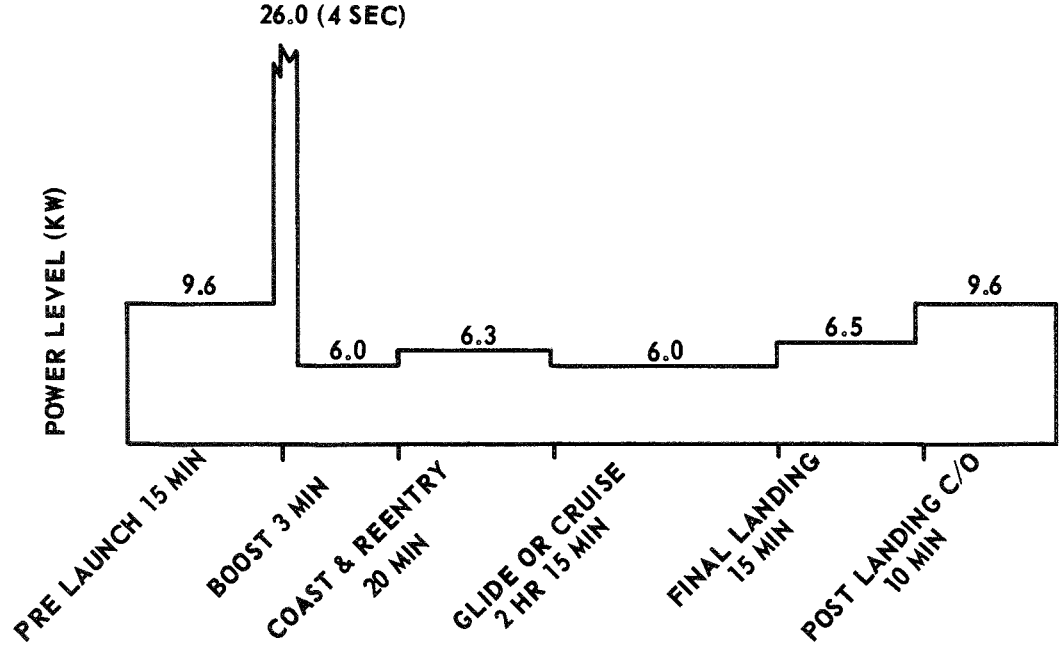


Figure 3-1. Space shuttle power profile (booster).

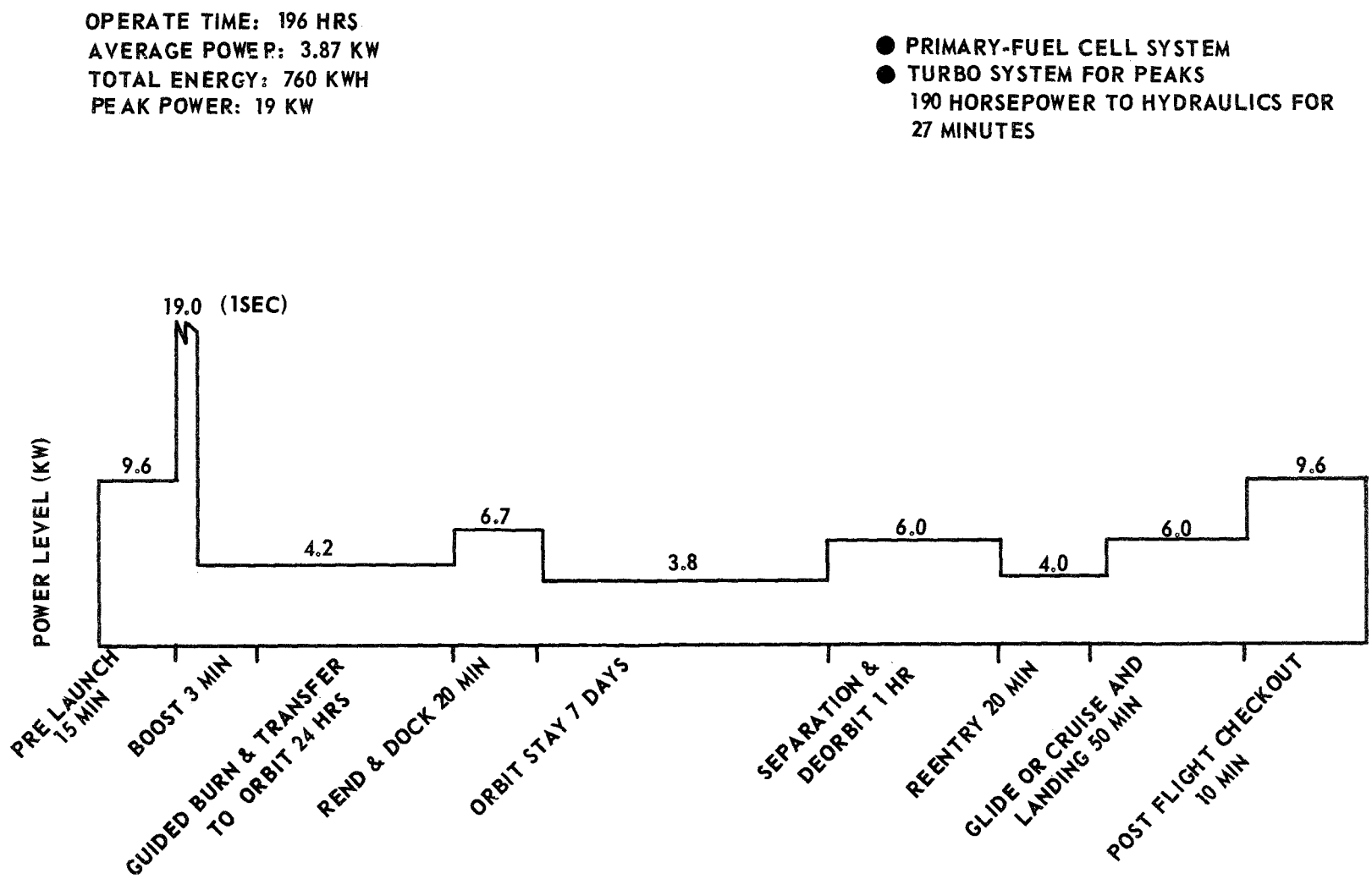


Figure 3-2. Space shuttle power profile (orbiter).

3.3 Power Sources Selection

To fulfill the shuttle power requirements, the following conclusions were reached: three power sources are desirable for the orbiter; the primary power source for normal operation should be a fuel cell; a back-up power system of batteries is needed to meet redundancy and safety requirements. For the short-term high-peak requirements during reentry and landing, a chemically-fueled turbine-driven system was selected.

It has been determined from the power systems requirements that two power sources are adequate for the booster. The primary power source for normal operation should be a battery system. Redundancy and safety requirements can best be met by additional batteries. The short-term high-peak requirements during reentry and landing are supplied by a chemically-fueled turbine-driven system. The principal reasons for selecting a primary power source of batteries for the booster are (a) mission time, (b) availability, (c) greater landing safety, (d) replaceability, and (e) lower cost.

The main reason for selecting different primary power sources for the booster and orbiter is the length of the mission. Careful examination of Figure 3-3 will reveal the reasons for this selection; for example, the primary battery weight for an 80-hour mission is approximately 5 times the weight required for a fuel cell.

The selected components for the booster and orbiter electrical power systems are shown in Figures 3-4 and 3-5. The major components selected for the electrical power system and their functions for the shuttle are shown in Table 3-3. Power for the vehicle during atmosphere operations will be supplied from alternators driven by the turbojet engines.

3.4 Analysis for Component Selection

The primary power sources analyzed in these trade studies were batteries and fuel cells. The principal parameters governing selection were weight and volume. Factors such as development status, reliability, cost, and servicing requirements were considered.

3.4.1 Batteries

Three primary battery types were considered as candidates for the shuttle power system: nickel-cadmium, silver-cadmium, and silver-zinc.

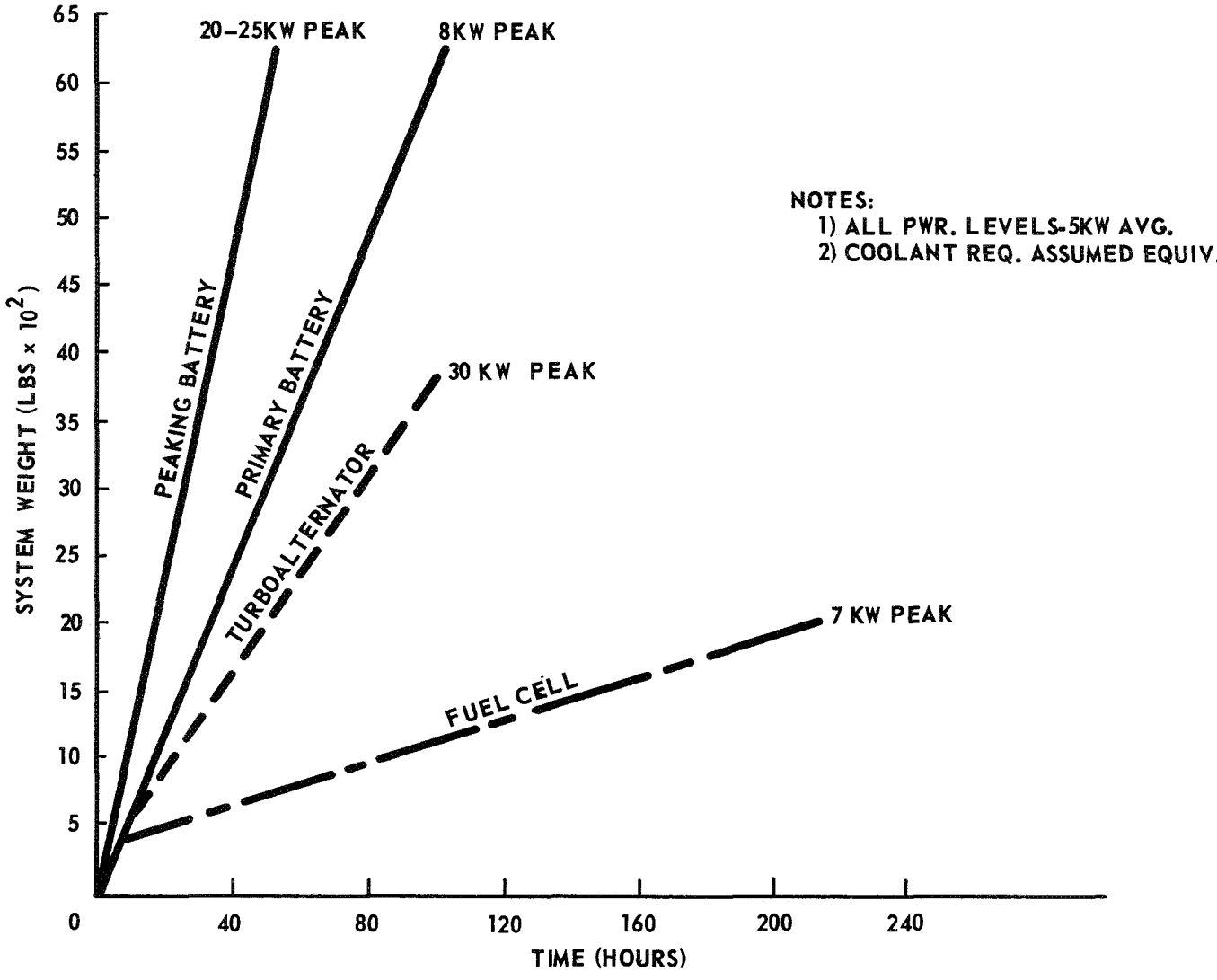


Figure 3-3. Power system relative weights.

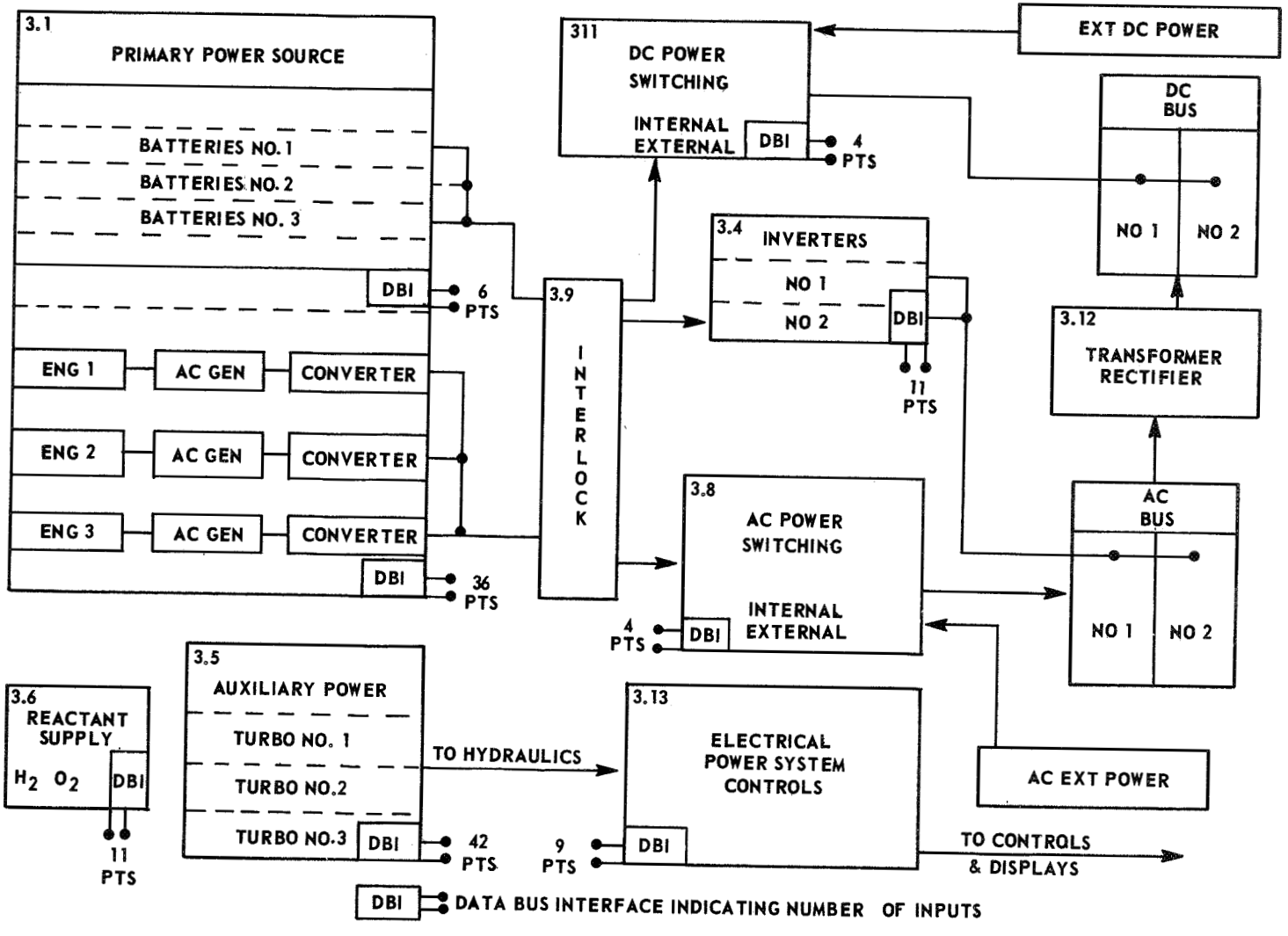


Figure 3-4. Electrical power system block diagram (booster).

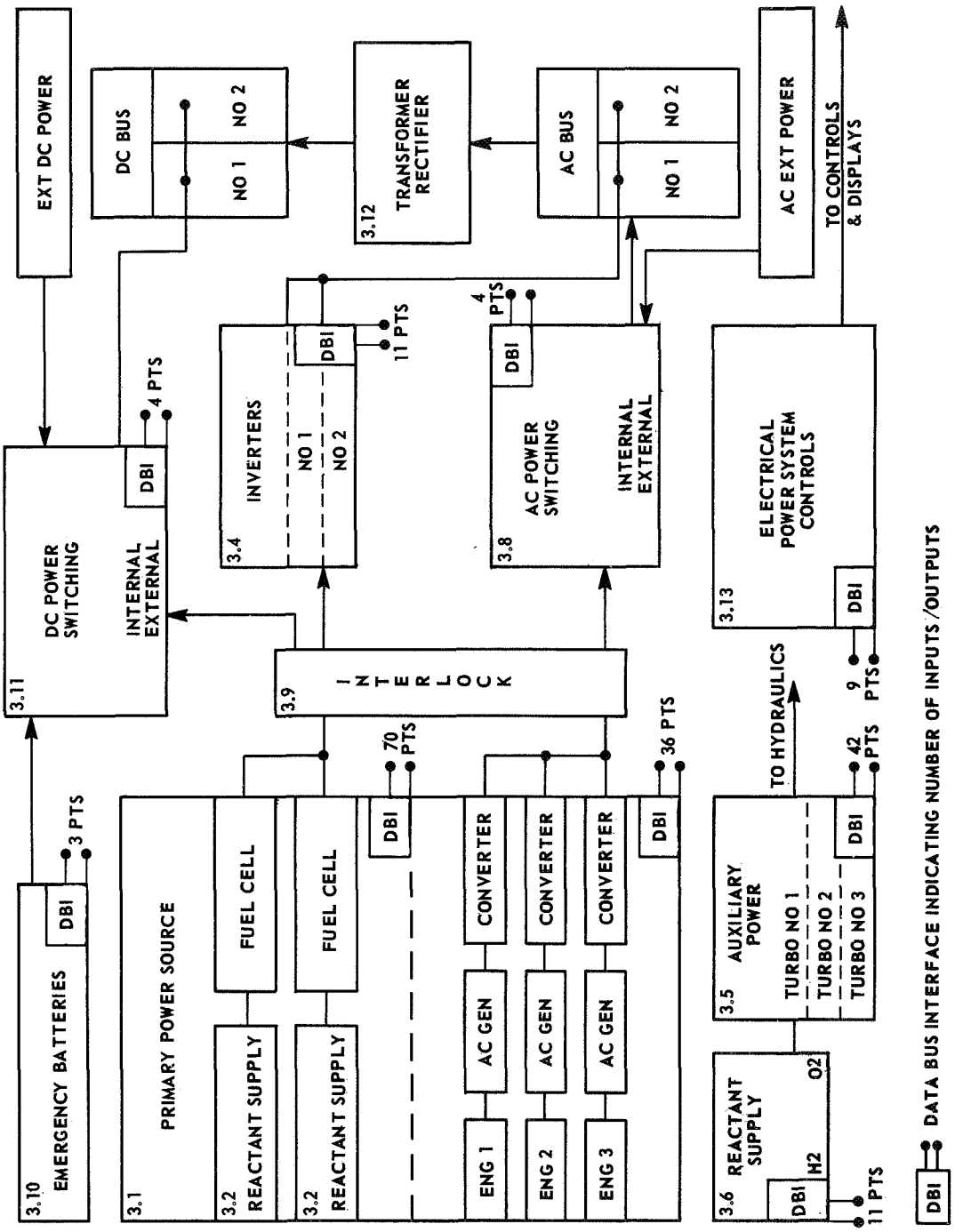


Figure 3-5. Electrical power system block diagram (orbiter).

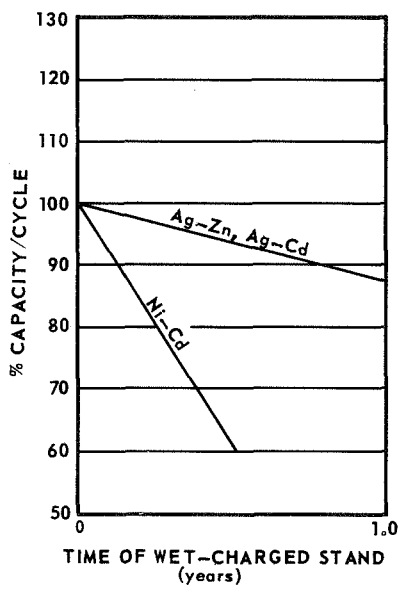
TABLE 3-3. SELECTED EPS COMPONENTS

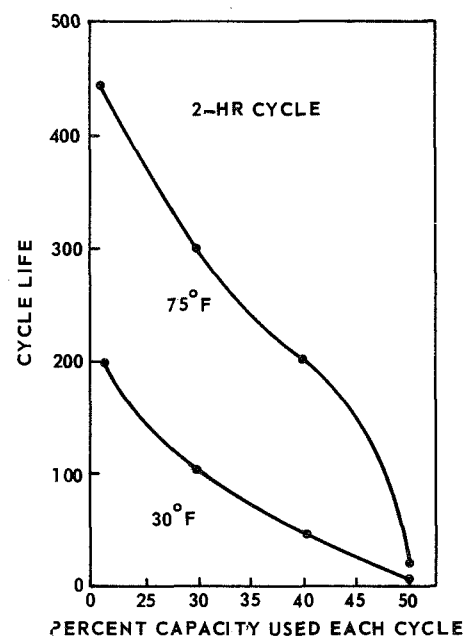
Function	Components	
	Booster	Orbiter
Primary power	Batteries, Silver-zinc	Fuel cells
Emergency power	Batteries, Silver-zinc	Batteries, Silver-zinc
Conversion	Solid-state inverters 3 ∅, 400 Hz, 115 volts (2 each - 2 redundant units)	
Distribution	dc, 28 ± 3 volts, two buses ac, 115 ± 5 volts 3 ∅, 400 Hz, two buses	

Silver-zinc is believed to have the most desirable characteristics. Typical characteristics of these batteries are shown in Figure 3-6. The energy density of the silver-zinc battery is significantly higher than for the other units, resulting in a weight and volume advantage. The nickel-cadmium battery has the best recharge capability for all systems studied. The silver-zinc and silver-cadmium batteries have a much longer wet-stand life capability. Since the shuttle has a 10-year life requirement, it was concluded that recharge-ability is not a significant factor since battery life is limited after initial activation. The higher energy density favors the Ag-Zn (30-75W-hr/lb) battery; therefore, it was selected for the conceptual design.

The primary battery power system for the booster was sized by the following procedure. From Figure 3-1, the ascent, descent, and reentry phases total 30 minutes and require power at a 6-kw rate. All the power required for these phases of the mission will be battery supplied. The power required from reentry to landing, a time span of 2.5 hours, is at a 6-kw rate. Approximately 50 percent of this load was considered critical and should be battery supplied. The total amount of power required for the above conditions is 10.5 kw-hr; however, to meet the reliability requirements, the system was sized for 15.75 kw-hr. Using 60 watt-hours per pound for silver-zinc, 265 pounds of batteries are needed for the booster. Seven standard batteries weighing 40 pounds each will meet this power requirement.

BATTERY TYPE	ENERGY DENSITY (W-H/LB)
NICKEL - CADMIUM	10-15
SILVER - CADMIUM	25-30
SILVER - ZINC	30-75





CYCLE-LIFE PERFORMANCE, EFFECT OF DEPTH OF DISCHARGE AND TEMPERATURE.

Figure 3-6. Typical battery characteristics.

The orbiter emergency power system, which consists of batteries, provides for a safe return in the event that both fuel cells fail. This emergency system was sized in the following manner. The power profile in Figure 3-2 reflects a time of approximately 2.5 hours from deorbit to landing. The electrical power required for a safe return was assumed to be at a 4-kW rate. The normal loads plus additional redundant requirements amounts to a total requirement of 15 kW-hr. A value of 60 watt-hours per pound was used in sizing the system, and approximately 250 pounds of batteries are needed for emergency power. Six standard batteries weighing 40 pounds each will meet this power requirement.

3.4.2 Fuel Cells

The fuel cell system converts chemical energy into electrical energy using techniques that avoid the thermodynamic limitations on efficiency imposed by the carnot cycle. Energy densities in the order of 15 to 25 W-hr/lb are achievable. The fuel consumption is about 0.79 to 0.91 lb/kW-hr for H_2 - O_2 systems. Fuel cell efficiencies are in the order of 50 to 65 percent.

There are three major fuel cell developers: General Electric (GE), Pratt and Whitney (P&W), and Allis Chalmers (AC). The AC and P&W approaches employ essentially the same technology, the main difference being the way in which by-product water is removed. Both approaches use a potassium hydroxide (KOH) impregnated asbestos matrix contained between the electrodes of the cell. AC uses a static water removal technique which employs a second KOH impregnated asbestos matrix spaced away from the H_2 electrode. The KOH concentration difference between the matrices causes water to diffuse from the cell and is removed through a low pressure system maintained on the back side of the matrix. P&W's approach causes water removal from the cell by a recirculating H_2 stream. This approach results in higher parasitic power loss. The best efficiency and voltage regulation was exhibited by the AC cell with its static water removal system. GE has a different technology using a solid electrolyte composed of a sulfanic acid ion exchange membrane. Water removal is accomplished by a wicking process to a manifold and then drawn off. Only the AC and P&W systems were considered in this design because of the large power requirements. Figure 3-7 and Table 3-4 illustrate the comparative data for these two fuel cells.

All the fuel cells considered operate in the current density range of 100 to 200 amperes per square foot. Considerable effort is being expended to raise this level to as much as 4000 amperes per square foot. Present technology permits fuel cell systems with 30 lb/kW weight, including all supporting accessories. The reactant weights for hydrogen and oxygen are 15 and 17 lb/kW, respectively.

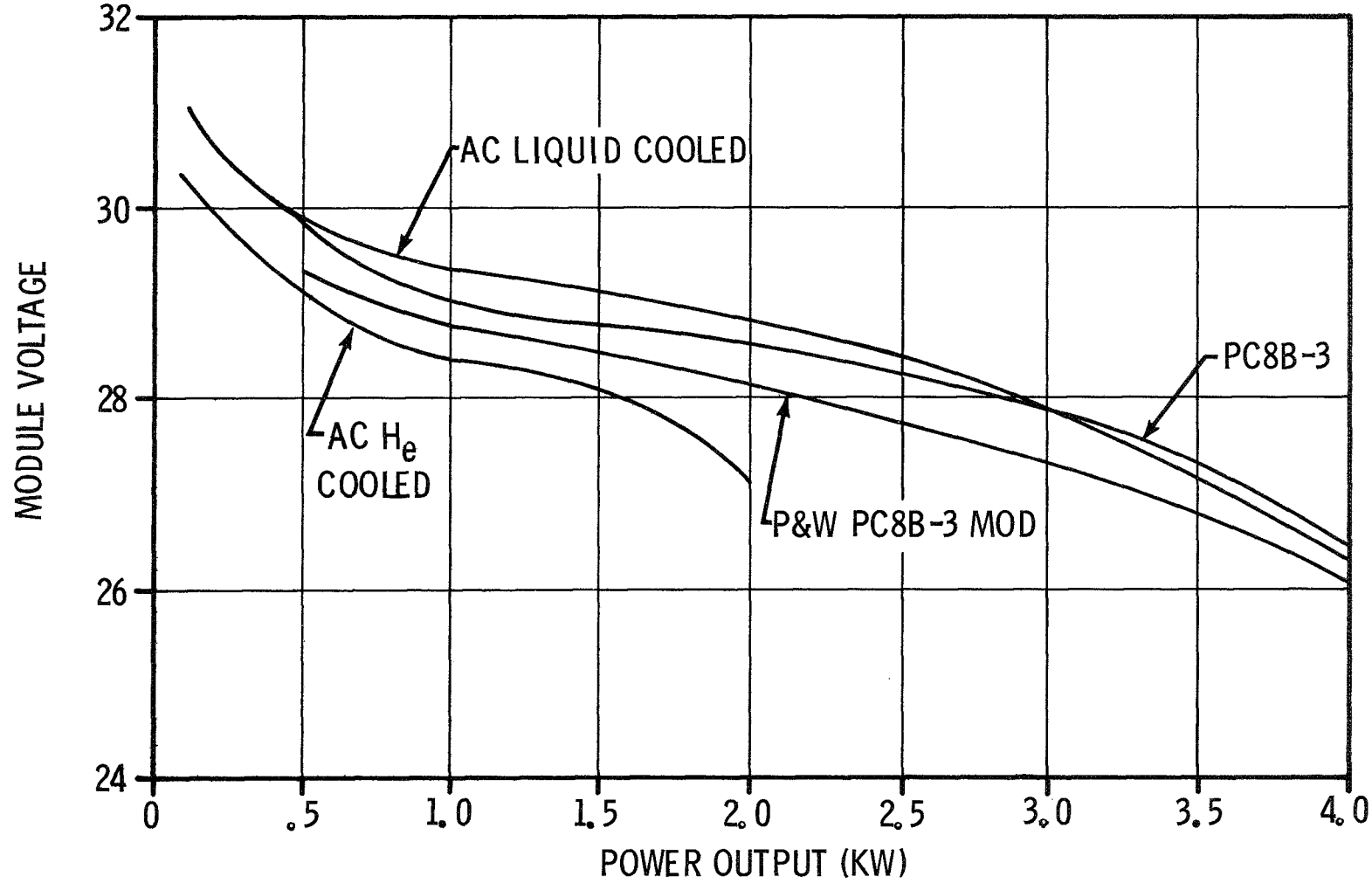


Figure 3-7. Fuel cell power and voltage comparisons.

TABLE 3-4. FUEL CELL SYSTEMS COMPARATIVE DATA

Type	AC		P&W	
	Helium Cooled	Liquid Cooled	PC8B-3	PC8B-3 Modified
Power (kW)	2.0	5.0	4.5	4.5 to 20*
No. of cells	31	30	31	31
Cell area (sq ft)	0.4	1.0	0.5	0.5
Fuel consumption (lb/kW)	0.8@1kW	0.8@ 1 kW	0.86 @ 1.4 kW	0.86 @ 1 kW
Max sustained load (kW)	3.2	5.0	4.5	*20 (1 hr)
Radiator required	Yes	Yes	Yes	Yes
System weight (lb)	190	170	105	132
Design life (days)	62	125	90	90
Operating temperature (°F)	195	195	190	190
Life test (hr)	3000	None	2100	None
Cumulative test (hr)	35 000	None	8100	None
Dimensions (in.)	35 x 21 x 16	11 x 11 x 32	18 x 18 x 20	20 x 20 x 18

For the fuel cell system to produce 3800 watts continuously for 192 hours and meet the failure criteria, two units should be operated in parallel. Emergency battery power is provided in the event both fuel cells should fail. Although the fuel cells are operated in a load-sharing mode, each fuel cell and the required reactants must be provided for full capacity of the entire mission.

It was concluded from data shown in Table 3-4 and Figure 3-7 that the AC liquid-cooled unit had the most desirable characteristics for this application. The voltage regulation of this system will adequately meet vehicle requirements without additional regulation equipment. Operating in the 1.5- to 2.5-kW range, the unit has better than ±1 volt regulation. From past experience, the vehicle requirements will probably be specified as 28 volts ±5 percent. For the two systems considered, the liquid-cooled unit exhibited the best fuel consumption rates. Since all fuel cell systems degrade from plate contamination with operation, the AC unit should operate longer for the same level of degradation. Also the large area will permit the AC unit to operate with higher peak loads at a lower current density per cell. The slightly lower initial weight for the P&W system will be compensated for by a lower reactant weight requirement of the AC unit.

The fuel cell system can be expected to have an internal resistance of 0.05 ohm. The load transient response time will be in the order of 25 microseconds. Normally, a 30-minute warm-up time will be required before the stack reaches 170° to 200° F and full load can be applied. Limited operation at temperatures well below their design point is possible. The power producing capability for one fuel cell is in the order of 60 watts, at a temperature of 0° F. For this condition the power to monitor, to control, and to supply other parasitic loads may require the total output of the system. These parameters will apply to both AC and P&W systems.

3.4.3 Auxiliary Power Unit (APU)

The auxiliary power unit on the shuttle vehicle provides the short-term peak demands not met by the electrical power system. The principal short-term demands are the requirements for hydraulic power to actuate the aerodynamic control surfaces during reentry and landing. Additional hydraulic power is needed for deployment of landing engines and extending the landing gear.

An APU is necessary on both the booster and the orbiter to meet these requirements. A separate propellant pressurization system will be

required to provide proper quantity and pressure of reactant to the APU on the booster. The system for the orbiter can be integrated with the attitude control system to provide proper quantity and pressure (100 psi) of reactant for the auxiliary power unit. The propellants (O_2 and H_2) will be supplied by the main propulsion residuals. It is believed this can be accomplished with little added weight penalty. A block diagram of the proposed auxiliary power system is shown in Figure 3-8.

3.4.3.1 APU Hydraulic Power Requirements

Three APU's are provided for each vehicle and each drives an independent hydraulic system. All three units provide power to the primary flight controls. Only two of the systems are connected to secondary controls requiring hydraulic power. The primary system provides hydraulic power to the elevons, ruddervators, and spoilers. The trailing-edge flaps are considered secondary controls and are connected to only two of the three independent hydraulic systems. The hydraulic power required by orbiter and booster is listed in Table 3-5. The operating times shown in Table 3-5 are considered adequate to supply hydraulic pressure required during the mission.

TABLE 3-5. APU HYDRAULIC POWER REQUIREMENTS, TWO STAGE SYSTEM

	Booster	Orbiter
Horsepower	567	190
Operating time (min)	6	27
Energy (hp-min)	1260	1680

Each hydraulic subsystem was sized to provide 50 percent full hinge moment at full rate to meet fail-operational[2], fail-safe criteria.

3.4.3.2 APU System Description and Selection

Turbomachinery systems using hydrazine blend fuel and H_2-O_2 fuel were analyzed. Either system is considered adequate for this operation. Both systems have very similar working characteristics. The H_2-O_2 system

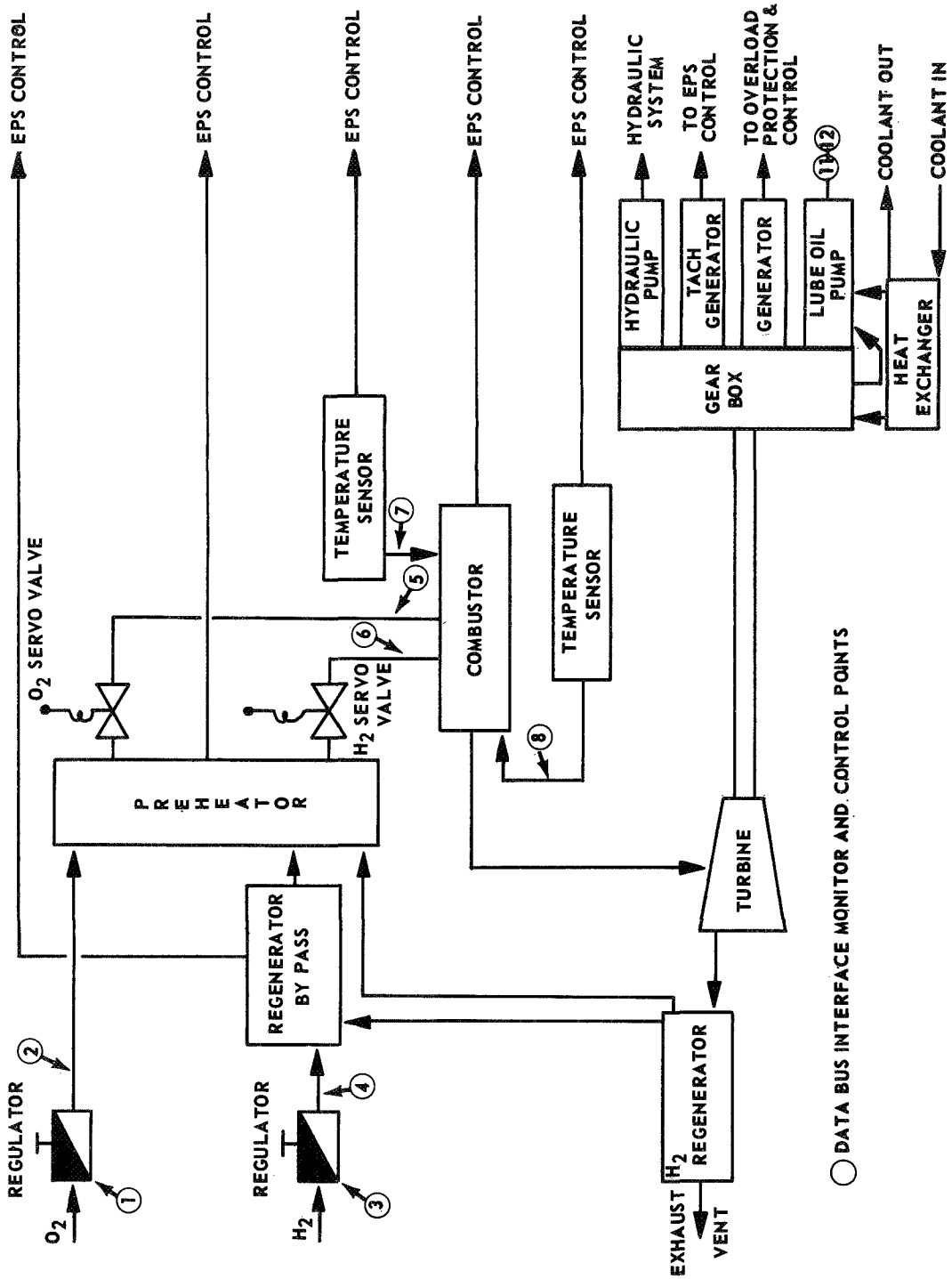


Figure 3-8. Auxiliary power source.

was selected because the main propulsion residuals could supply the fuel with little or no added penalty. Also the H_2-O_2 system has two to three times better fuel-consumption-to-horsepower-output ratio. The H_2-O_2 system will be approximately 100 pounds lighter for the orbiter than the hydrazine-fueled system. The comparative weights for the two systems are given in Table 3-6. The H_2-O_2 system should be more reliable when starting in extremely cold conditions. It would require a special blend of hydrazine to prevent freezing at - 65°F.

TABLE 3-6. APU WEIGHT SUMMARY

System Data	Booster	Orbiter
Hydrazine		
Turbine unit (lb)	223	75
Fuel tank	53	49
Fuel	202	245
Total weight (lb)	478	369
H_2O_2		
Turbine	267	94
Regenerator	53	19
Hydrogen	52	78
Oxygen	42	63
Total weight (lb)	414	254

One possible fuel for this application is Sunstrand 70-20-10, a blend containing hydrazine, monomethyl hydrazine, and hydrazine nitrate. A solid propellant ignition system will be necessary to preheat the decomposition chamber prior to fuel injection, to accelerate the turbine, and pressurize the fuel tank to provide inlet pressure to the fuel pump. The H_2-O_2 system should have a low development risk since this unit was developed to the prototype stage

for the Dyna-Soar. Further development will be required to satisfy and demonstrate the reliability requirements for the shuttle vehicle. A block diagram of the proposed oxygen-hydrogen system is shown in Figure 3-8.

3.5 Power Conditioning and Distribution

The principal distribution system for the orbiter will be a 28-vdc system. The remaining loads will be supplied by a 115-volt, 3-Ø, 400-Hz system for special purposes, such as for the inertial measuring unit. During the aircraft mode of operation the primary power generation source will be an alternating current system. The large power requirements during the aircraft mode for motor-driven pumps and heating loads are more efficiently met with an alternating current system. Required dc loads can be supplied with a transformer rectifier system that is efficient, light in weight, and reliable.

3.5.1 Direct Current System

The principal reasons for selecting a 28-vdc distribution system were low-level current loads and availability of man-rated flight components. The majority of the circuits for the booster and orbiter require less than 5 amperes of current, as shown by the examples in Table 3-7. It is believed that this level of power can be met without excessive penalty in cable weight. There would be little or no development risk for breakers, contactors, and other items required in a 28-volt distribution system. A higher distribution voltage could conceivably result in some weight savings; however, the additional development, documentation, and required testing are believed to involve a greater risk than the weight penalty.

Electrical components are usually manufactured with 30- and 120-volt ratings. The state-of-the-art in power transistor development technology allows a maximum collector sustaining voltage of 250 volts. Utilizing the best-known power switching techniques from a 120-volt distribution system, the transistor would see 240 volts peak at the collectors. These power inversion systems inherently produce some spikes on the voltage waveforms. For this reason, a 120-volt distribution system is considered to have questionable reliability. A tap at some intermediate point for the inverters is a possible solution that may result in a weight saving.

To connect four fuel cells in series to obtain a 120-volt distribution system would also have questionable reliability. Manufacturing a single unit with a 120-volt rating would increase the complexity of the stack and increase the voltage insulation problems. Because of the above problems a 28-volt distribution system was selected.

TABLE 3-7. TYPICAL DISTRIBUTION LOADS 28-VDC SYSTEM

Function	Amperes
Autopilot, master warning, and control	10
Position lights (external)	20
Duct lip position and ice detection	1
Valve controls	5
Landing gear and brake warning	10
Fuel and oil temperature transducer	1
Emergency pressurization	1
Anti-ice control and indication	2
Flap control	2
Anti-skid control	20
Windshield anti-ice control	1
Structural overheat warning	1

3.5.2 Alternating Current System

An alternating voltage for distribution can easily be stepped up or down by known transformer techniques. Increasing the voltage permits the use of lighter transmission cable for the same power level. If the cable run is short, the cable weight saving may be offset by the additional weight required for step-up and step-down transformers.

Changing of the fundamental frequency from 400 to 2000 Hertz theoretically could reduce the transformer weights for step-up and step-down operations. The component weights for servosystems, platforms, motors, and equipment transformers could also be reduced, theoretically, by a factor of 5; however, the reactance (X_L) would be increased by 5 times. Larger transmission lines and different impedance-matching techniques would be required to keep the voltage drops in the 2000-Hertz system at the same level of those in the 400-Hertz system. Radio frequency interference problems would be expected to increase in the 2000-Hertz system. The primary reason for selecting the 400-Hertz for the AC distribution system was that nearly all guidance systems hardware developed in this country for missiles and aircraft are at this frequency. The cost for high frequency component development, documentation, testing, and man-rating would overshadow the possible weight savings.

3.6 Shuttle EPS Weight and Volume Estimates

The principal difference in the booster and orbiter systems is the primary source of batteries for the booster and fuel cells for the orbiter. Batteries are the emergency power source for both stages. The component weights and volumes for these systems, as shown in Figures 3-4 and 3-5, are given in the following paragraphs.

3.6.1 Batteries

Silver-zinc was selected because of its superior energy density of 75 W-hr/lb. Each battery weighs approximately 40 pounds.

3.6.2 Rectifiers

Isolating rectifiers located in the central power bus will weigh about 1.0 pound each when mounted. Eight rectifiers are required.

3.6.3 Circuit Breakers (dc)

The circuit breakers will be solid state devices with automatic as well as remote reset and trip capability. For each load, the breakers will provide an on-off status which can be programmed through the computer system. Each load has a separate breaker providing overload protection for equipment and wire runs from the power bus to the load. The estimated 150 circuits allow for some growth capability. Each breaker will weigh about 0.5 pound for a total weight of 75 pounds.

3.6.4 Generators (ac)

The booster and orbiter in the aircraft mode of operation will have ac loads supplied by a cycloconverter. An ac generator is driven directly by the main engine. The cycloconverter can accept a variable shaft rpm input, and deliver a constant-frequency three-phase, 400-Hertz, 115/200-volt output utilizing special wave-shaping techniques. The generator weighs about 50 pounds, the cycloconverter 50 pounds, and the oil coolant loop about 10 pounds. The system has an electrical output rating of 40 kv-A.

3.6.5 Transformer Rectifier

The transformer rectifier system supplies the required 200-ampere 28-vdc loads for the aircraft mode of operation. These units weight approximately 20 pounds each, including transformer, rectifiers, and mounting brackets.

3.6.6 Wiring Estimates

The wire sizes and weights to install the selected components for the shuttle system have been estimated. The fuel cell requires about 45 feet of number 5 gauge stranded cable, which weighs approximately 15 pounds. including harness, feed-throughs, and connectors.

The generator installation will require approximately 120 feet of bus feeders, which weigh about 100 pounds including supports, feed-throughs, and connectors.

The battery installation for the booster will require about 40 feet of number 5 gauge stranded cable, which weighs about 12 pounds including harness and mounting brackets. The orbiter installation will require about 40 feet of number 6 gauge stranded cable which weighs about 10 pounds, including harness and mounting brackets.

3.6.7 Fuel Cell

A liquid-cooled 5.0-kw fuel cell system was selected. The normal system load will be 3870 watts continuously. Two units are used to meet design reliability requirements and provide overload and growth capability. During normal operation both cells will share the load; however, reactant supply is adequate for a single unit to accomplish the entire mission. The weights for the fuel cell system are given in Table 3-8. The radiator weight and volume for the fuel cell are not included in this table.

TABLE 3-8. FUEL CELL SYSTEM WEIGHTS AND VOLUMES

Item	Weight (lb)	Volume (ft^3)
Fuel cells (2 units)	340	3
Oxygen	1200	
O_2 tanks	275	30
Hydrogen	150	
H_2 tanks	300	50
Valves, heater lines	100	2
Total	2365	85

3.6.8 EPS Weight and Volume Summary

The electrical power system components for the booster and orbiter are summarized in Table 3-9. APU weights should be carried under subsystem 1.0, Structural/Mechanical.

3.7 Data Bus Interface

A detailed analysis was made for each component of the EPS to determine the number of data bus interface connections. Each component will require a signal conditioning unit that will transform the information so that it can be accepted by the data bus. The signal conditioning units and the number of test points are shown for the EPS in Figure 3-4 and 3-5. A complete list giving function, type of measurement, etc., for the EPS is furnished in Appendix A, Section 3.0. As an example, Test Point No. 13, shown on the fuel cell schematic in Figure 3-9, is defined in Section 3.0 of Appendix A. This test point is a sensor signal of the stack temperature, analog in nature, has a range of 150° to 210°F, a resolution ±2°F (monitored signal), and requires an alert when the stack temperature exceeds 200°F. All test points are numbered, named, and located as shown on the diagrams for each respective system. The total number of test points for any power source is less than 100 measurements. The EPS data bus inputs are status type measurements and are usually called up on demand. Only when some system is performing

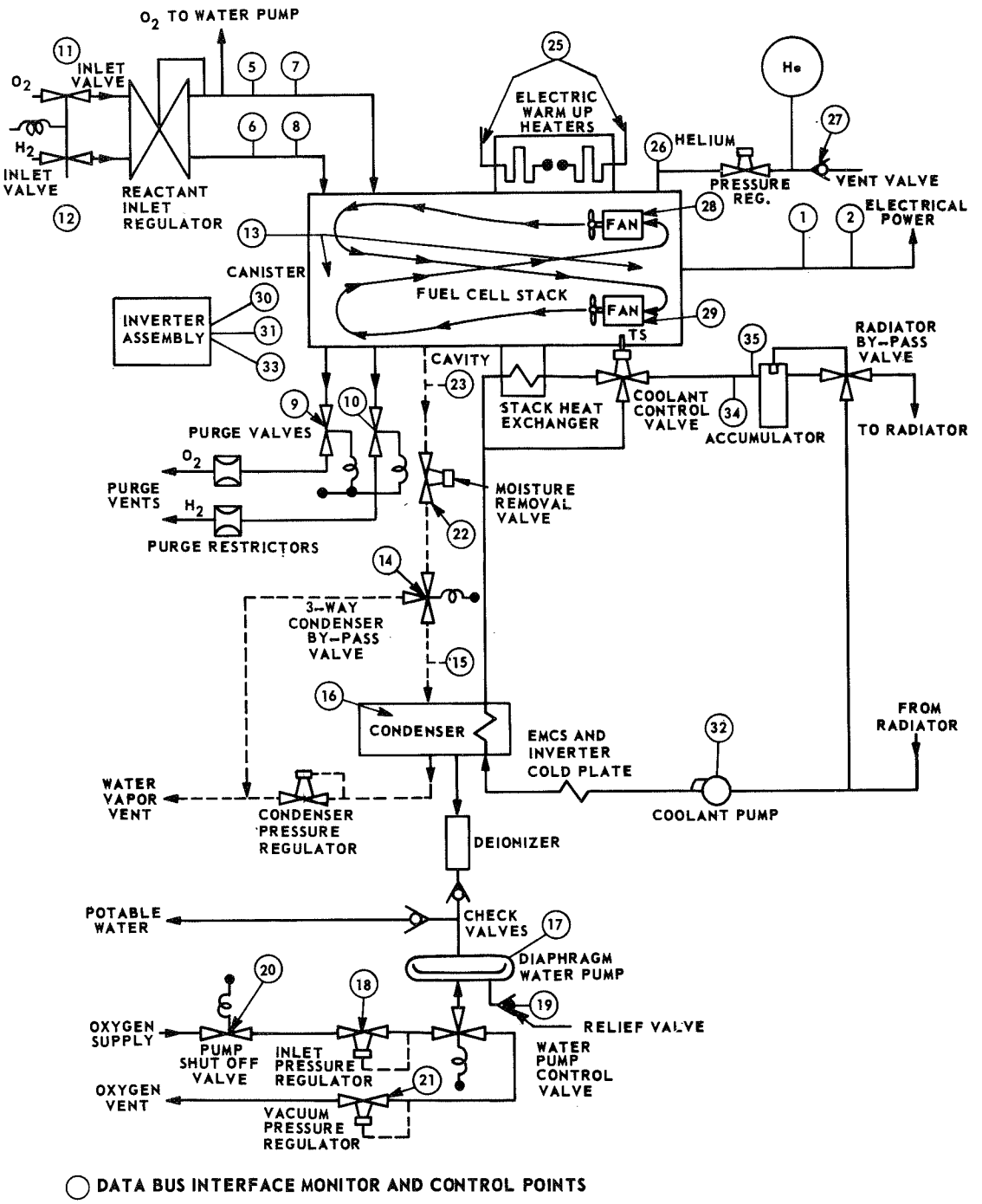


Figure 3-9. Fuel cell module schematic.

TABLE 3-9. EPS WEIGHT AND VOLUME SUMMARY

Item	Booster		Orbiter	
	Weight (lb)	Volume (ft^3)	Weight (lb)	Volume (ft^3)
Fuel cell system			2365	85.0
Silver-zinc batteries	400 (10 ea)	7.0	240 (6 ea)	4.0
Ac generators	150 (3 ea)	6.0	150 (3 ea)	6.0
Converters	150 (3 ea)	4.0	150 (3 ea)	4.0
Generator coolant	30 (3 ea)	0.5	30 (3 ea)	0.5
Transformer rectifier	60	1.0	60	1.0
Isolation rectifiers	8	0.5	8	0.5
Dc circuit breakers	72	1.0	72	1.0
Wiring	112	2.0	127	2.0
Ac breakers	42	1.0	42	1.0
Bus enclosure	20	2.0	20	2.0
Total	1044	25.0	3264	107.0

improperly would an automatic warning be issued. The information will be available to the data bus at all times, but the primary usage will be during the preflight and post-flight checkout periods. These interrogations will be to determine that the various systems are working properly. The small number of test points and the low data rates will not be the determining factor in the sizing of the memory capability and speed of the computer system.

4.0 Environmental Control

Considerable information and data on the environmental and life support subsystem (EC/LSS) of the space shuttle vehicle have been generated by various contractor studies [1, 2, and 3] as well as by in-house NASA investigations during the Phase-A ILRV studies. This available reference material was freely used to determine the necessary background and interface information to permit the IAS design to be pursued. As an example, the EC/LSS as generated by LMSC is illustrated by Appendix D, 4.0. Primarily, requirements, as well as constraints, imposed on the IAS by the EC/LSS of the various configurations were identified and utilized where necessary in the respective IAS subsystem studies and investigations. The contractor EC/LSS data were reviewed by PD-DO-ME; comments from this review [5] are reflected in the EC/LSS portions of Appendices A and D.

5.0 Guidance, Navigation, and Control Subsystem

This section identifies, describes, and provides the rationale for a conceptual design of the shuttle guidance, navigation, and control subsystem. Common basic terminology has been established for use throughout the program.

5.1 Fundamental System Definitions

Coordinate systems, symbols, and polarity definitions are basic to the development of vehicle equations. These equations describe internal vehicle operations and vehicle dynamics, and form the mathematical foundation for analysis and simulation. They affect hardware conventions and, to some extent, even the manner in which functions are performed. The use of standard terminology improves communication and working efficiency throughout the program. Therefore, it is important to start with simple mathematically consistent terminology which is widely recognized in the aerospace field. Unfortunately, the shuttle vehicle is a combination of rocket and aircraft technologies which are not entirely compatible with regard to nomenclature.

The terminology selected for the shuttle has been based on the following ground rules:

(1) All coordinate frames and positive rotation about these frames follow right-handed convention.

(2) Derivatives of functions with respect to time follow standard dot notation.

(3) Greek letters designate angular functions.

(4) Subscripts of angles indicate axis of rotation and subscripts of coordinate axes designate the reference frame.

(5) Standard aircraft and rocket terminology is retained when possible.

Several basic coordinate frames which are used for control and dynamic equations are defined in Table 5-1. The relationship between frames is clarified in Figures 5-1, 5-2, and 5-3. The body-fixed frame, as defined, is universally recognized in both the aircraft and space fields. The orientation of the accelerometer set relative to the body set has not been determined. The

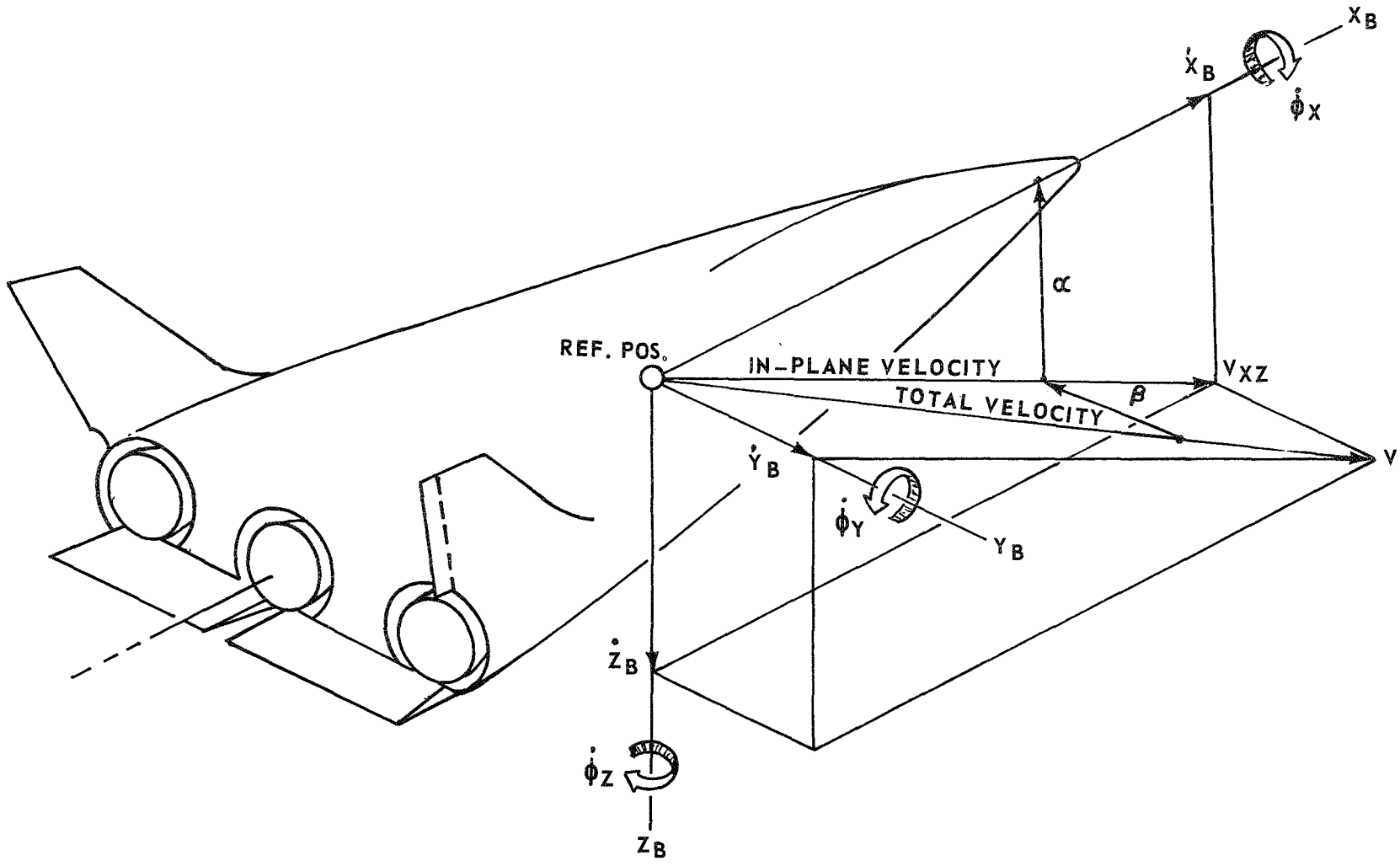


Figure 5-1. Body-fixed axis system.

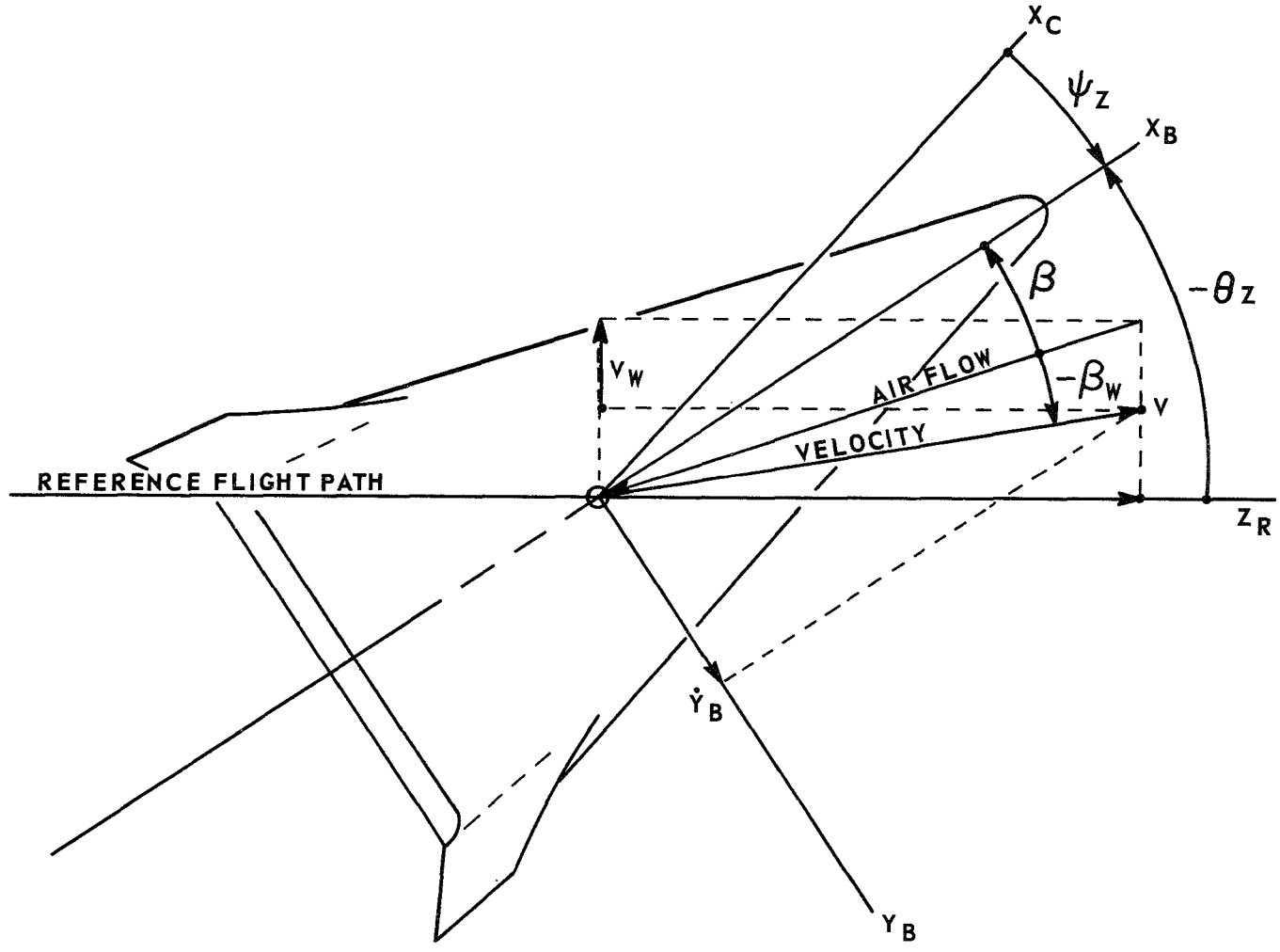


Figure 5-2. Angular relations for lateral dynamics.

TABLE 5-1. COORDINATE DEFINITIONS

X_A, Y_A, Z_A	Accelerometer coordinates: The axes are aligned with the accelerometer measuring direction. The orientation relative to body coordinates has not been selected.
X_B, Y_B, Z_B	Body-fixed coordinates: The X axis lies on the long axis, positive toward the nose. The Y axis is positive to the pilot's right and the Z axis completes the triad.
X_C, Y_C, Z_C	Command system: These coordinates define the commanded or referenced orientation of the vehicle.
X_E, Y_E, Z_E	Space-fixed, earth-centered coordinate system with X_E along line of nodes of equator and ecliptic (vernal equinox) and Y_E along north pole
X_L, Y_L, Z_L	Local coordinates, origin on surface of earth, X_L along local north, Y_L local vertical
X_N, Y_N, Z_N	Navigation Coordinates, ideally aligned along $X_E Y_E Z_E$
X_P, Y_P, Z_P	Principle axes: The origin is at the center of mass with the axes along the principle axes of inertia.
X_{rw}, Y_{rw}, Z_{rw}	Runway coordinates, X_{rw} along runway centerline, Y_{rw} along local vertical, origin at intersection of glide slope and runway
X_R, Y_R, Z_R	Radial system: The origin is at earth's center with the X axis through the instantaneous center of mass of the vehicle. The Z axis is in the flight plane.
X_S, Y_S, Z_S	Space-fixed system: The origin is at earth's center and the X axis is aligned to the body fixed X axis at the time of initialization (immediately before lift-off and reentry). The Z axis is in the flight plane.
X_W, Y_W, Z_W	World coordinates, earth-fixed centered with X_W through Greenwich Meridian, Y_W through north pole.

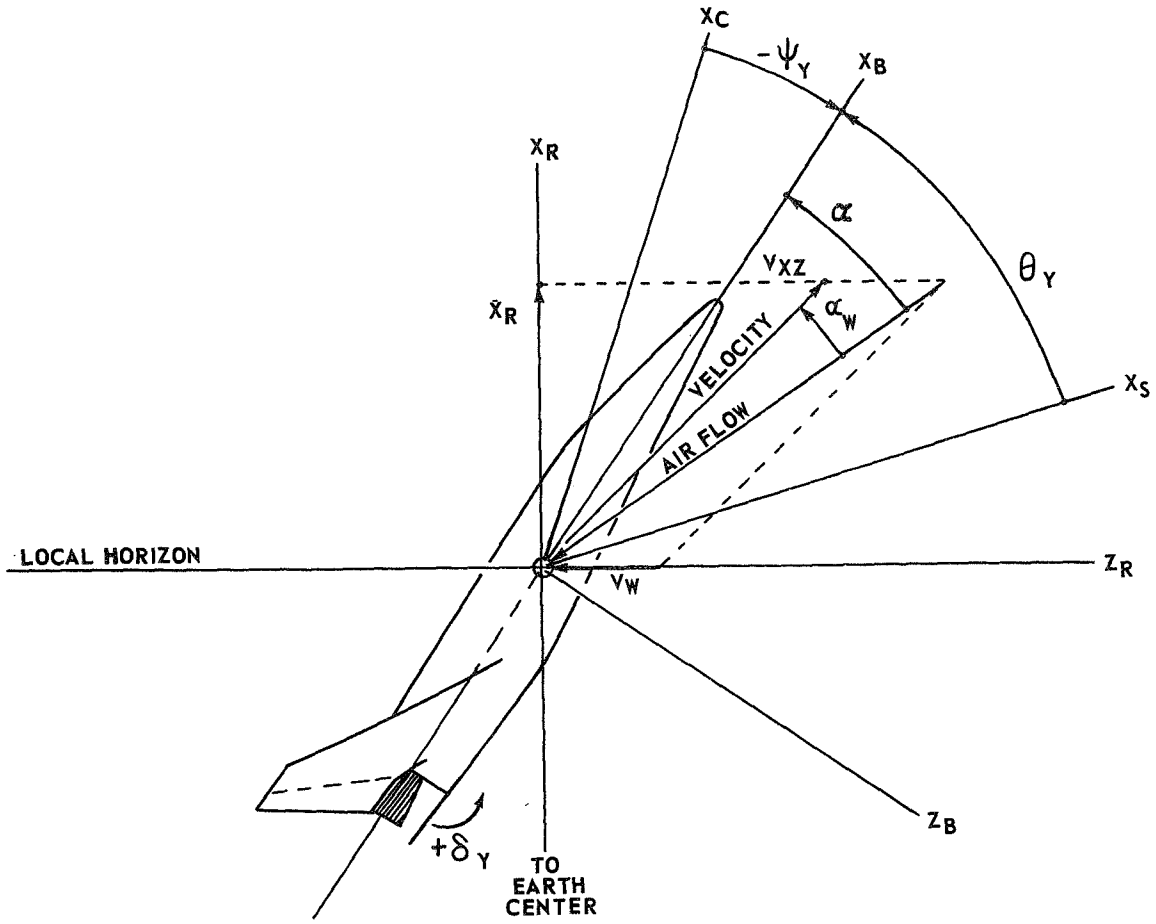


Figure 5-3. Angular relations for longitudinal dynamics.

alignment of the other frames is based on initialization to the body set at some particular time near lift-off and reentry.

The selection of symbols is complicated because those in common usage at MSFC are not entirely compatible with those used in the aircraft industry. The symbols which have been chosen are a compromise between the two. They conform to the ground rules, are compatible with the Marshall Vehicle Engineering System (MARVES) definitions and are only slightly different from aircraft terminology. Symbols are defined in Table 5-2 and many are shown on the figures. Typical stability derivatives that are commonly used are illustrated by Table 5-3.

The aircraft industry used English units exclusively; however, considering the distinct advantages of the metric system, all aerodynamic, navigation, and control studies are being carried out using the MKS International System of Units.

TABLE 5-2. DEFINITION OF SYMBOLS

Symbol	Units	Definition
A	deg	Azimuth
A_o	—	Attitude gain
A_1	s	Rate gain
A_R	m^2	Reference area of lifting surface
α	deg	Angle of attack (X-Z plane)
b	m	Maximum span of lifting surface
β	deg	Angle of sideslip (X-Y plane)
β_W	deg	Apparent sideslip from cross wind
C_D	—	The coefficient of drag
c	m	Mean aerodynamic chord
χ	deg	Guidance orientation commands
δ	deg	Actual actuator position
$\dot{\delta}$	deg/s	Actual actuator rate
δ_C	deg	Actuator commands
E_a	m	Elevation of point a above sea level
F	N	Total force on vehicle
F_A	N	Aerodynamic force

TABLE 5-2. (Continued)

Symbol	Units	Definition
F_G	N	Gravitational force
F_T	N	Thrust force
g	m/s^2	Acceleration of gravity
g_o	m/s^2	Acceleration of gravity at sea level
γ	deg	Angle which relates space-fixed coordinate system to radial coordinate system.
γ_p	deg	The angle between the velocity vector and the local horizontal
H	$N \cdot m \cdot s$	Total angular momentum
h	m	Altitide
I	$N \cdot m \cdot s^2$	Moments of inertia about body axes
I_p	$N \cdot m \cdot s^2$	Moments of inertia about principle axes
I_{xz}	$N \cdot m \cdot s^2$	Product of inertia
K_a	1/m	Atmosphere characteristic constant
L/D	—	Lift-to-drag ratio
M	kg	Mass of vehicle
$\dot{M}$	kg/s	Mass flow rate of fuel from vehicle
MACH	—	Mach number

TABLE 5-2. (Continued)

Symbol	Units	Definition
M_o	kg	Initial mass of vehicle
M_A	N• m	Aerodynamic moment
M_V	N• m	Total moments on vehicle
M_T	N• m	Thrust moment
ω	1/s	Natural frequency
p	N/m^2	Atmospheric pressure
$\dot{\phi}$	deg/s	Angular rates of the body in inertial space
ψ	deg	Attitude error angles
q	N/m^2	Aerodynamic pressure
R_E	m	Radius of the earth
R_s	m	Distance from center of earth to the vehicle
ρ	kg/m^3	Mass density of air
ρ_o	kg/m^3	Mass density of air at sea level
t	s	Time
θ	deg	Euler angles between body and reference axes
$\dot{\theta}$	deg/s	Euler angle rates between body and reference axes

TABLE 5-2. (Concluded)

Symbol	Units	Definition
V	m/s	Total velocity
V_A	m/s	Velocity of sound
V_W	m/s	Velocity of wind
W	N	Weight of vehicle

TABLE 5-3. AERODYNAMIC STABILITY DERIVATIVES FOR BODY COORDINATES

Symbol	Units	Definition
CA	—	Axial force coefficient for zero angle of attack
CA_α	—	Axial force coefficient for variation with angle of attack
CAq	—	Axial force coefficient variation with pitch rate
CY_β	1/rad	Lateral force coefficient variation with sideslip angle
CYr	1/rad	Lateral force coefficient variation with yaw rate
CYp	1/rad	Lateral force coefficient variation with roll rate
CY_δ	1/rad	Lateral force coefficient variation with rudder deflection
$CY_{\dot{\beta}}$	1/rad	Lateral force coefficient variation with sideslip rate
CN_α	1/rad	Normal force coefficient variation with angle of attack
CN_δ	1/rad	Normal force coefficient variation with elevon deflection
CNq	1/rad	Normal force coefficient variation with pitch rate
$C\ell_\beta$	1/rad	Roll moment coefficient variation with sideslip angle
$C\ell$ r	1/rad	Roll moment coefficient variation with yaw rate
$C\ell$ p	1/rad	Roll moment coefficient variation with roll rate
$C\ell_\delta$	1/rad	Roll moment coefficient variation with aileron deflection
Cm_α	1/rad	Pitch moment coefficient variation with angle of attack
Cmq	1/rad	Pitch moment coefficient variation with pitch rate
$CM_{\dot{\alpha}}$	1/rad	Pitch moment coefficient variation with angle of attack rate
Cm_δ	1/rad	Pitch moment coefficient variation with elevon deflection
Cn_β	1/rad	Yaw moment coefficient variation with sideslip angle
Cnr	1/rad	Yaw moment coefficient variation with yaw rate
Cn_δ	1/rad	Yaw moment coefficient variation with rudder deflection

5.2 Control System Requirements

This subsection identifies functions that must be performed by the control system. These functions are based on overall mission requirements as presently defined.

5.2.1 Booster

The boost vehicle must operate in two different configurations during its mission: (1) the launch configuration, which includes the orbiter, and (2) the return configuration, without the orbiter. In addition, there are various flight phases which introduce completely different control requirements. These flight phases can be identified in such a way as to isolate the control problems for a particular flight regime. They are (1) lift-off, (2) atmospheric boost, (3) separation, (4) coast, (5) reentry, (6) glide, and (7) landing maneuvers. Each flight phase contains unique problems which will require a control system design of sophistication and flexibility. The problems associated with each phase will be described in more detail in the following paragraphs.

Lift-off dynamics may be complicated by a launch configuration which has a large center-of-gravity offset in the pitch plane. This offset would result in both rotation and translation of the vehicle immediately after lift-off unless specific action is taken to prevent it. This motion is undesirable before clearing the launch support equipment.

The powered ascent phase does not appear to hold any unusual problems except for abort during high dynamic pressure and the fact that the lifting surfaces will make the vehicle more sensitive to winds than an ordinary cylindrical vehicle configuration.

Separation dynamics will be more critical than for stacked stages because the vehicles will be attached side by side. This creates greater chance for post separation contact and thrust impingement on the booster.

Attitude control using reaction control system (RCS) thrusters will be required to orient the booster for reentry. The proper mix of RCS and aerodynamic control during the beginning of reentry may be critical. Aerodynamic surfaces will be the primary means of control during the main reentry region. The vehicle must reenter with a large angle of attack (about 50 degrees) to provide the proper L/D. To achieve this attitude, the control surfaces must be near full deflection. The phugoid mode is very significant during reentry and must be damped to prevent large altitude excursions and efficiently remove orbital energy. The scheme for damping the phugoid mode should not

alter L/D appreciably, because reentry conditions will change. Pitch control is not a recommended solution; however, rotation of the vehicle about its velocity vector appears to be satisfactory.

The supersonic glide should not present any unusual control problems. There will probably be a maneuver following acquisition of ground-based beacons and the operation of jet engines may introduce some new disturbing torques. The transonic region, in contrast, may create serious control problems as a result of rapidly changing aerodynamic characteristics. Subsonic flight should not generate any new requirements unless deployable lifting surfaces (if used) change control characteristics significantly.

During landing, the vehicle must perform flare and decrab maneuvers. The flare maneuver alters the glide path to provide near null altitude rate at touchdown. The decrab maneuver removes any yaw bias resulting from disturbances such as cross wind so that the vehicle is aligned with the runway at touchdown. The sensor requirement is obviously critical for the landing phase.

5.2.2 Orbiter

The orbiter vehicle will maintain a given configuration with the possible exception of swing wings which could be deployed after reentry. The flight phases for the orbiter are (1) separation, (2) guided burn, (3) orbit, (4) reentry, (5) glide, (6) landing maneuvers, and (7) self-ferry.

Normal separation dynamics are as critical for the orbiter as the booster, and abort separation in the atmosphere may be more serious if the orbiter engines are not operating. There also exists the problem of control during engine startup.

Orbital operations will require RCS thrusters for both attitude and translational control. It may also be necessary to use two different thruster sizes to satisfy the performance requirements for both of these control functions. Rendezvous and alignment of the inertial system before reentry will require special sensors not carried by the booster.

The other flight phases, through landing, will be almost the same as for the booster; however, the orbiter will probably be required to take off from the landing field and fly back to the launch site using jet engines. This may introduce additional control requirements.

5.3 Navigation System Requirements

The fundamental requirements of a navigation system are (1) to provide sufficiently accurate position and velocity information for a guidance system to safely accomplish all mission phases, and (2) to provide sufficiently accurate attitude information for an attitude control system to safely control and stabilize the vehicle attitude during all mission phases.

A requirement for minimum dependence on ground equipment will impose further constraints on a number of navigation areas if this requirement is strictly observed. Prelaunch alignment must be accomplished by means of some type of earth-rate gyrocompassing. If the inertial navigation sensors are of sufficient quality, the gyrocompassing may be performed inside the computer; otherwise, a dynamic slewing may be required. Ground-based tracking updates are quite desirable for rendezvous and just prior to reentry; however, for rendezvous, the ground independence requirement will require a long-range target-tracking capability in addition to the short-range rendezvous and docking capability. Since a precise position update is required just prior to reentry, either an extremely precise horizon scanner (state-of-the-art horizon scanners are probably inadequate) or landmark trackers (this will require a man to select, verify, aim, and "shoot" the landmark) are required. The prereentry update complications would be greatly relieved by using ground-based tracking.

At the end of the reentry phase (about 30.5 kilometers or 100 000 feet), another position update will be required to locate the vehicle with respect to the landing field. For post-reentry and landing phases, use will be made of existing civilian and military radio navigation and landing aids.

The most stringent navigation requirements are on the orbiter which must have the capability of realigning a navigation coordinate frame that will drift significantly during the long mission time. The booster, since it is expected to have a relatively short mission time, may not require realignment during flight but will require a position update after reentry and possibly prior to reentry. Prelaunch and landing navigation requirements are the same for both the orbiter and the booster.

5.4 Control System Description

A reference control system design has been selected based on the system requirements. The purpose of this reference concept is to provide a starting point from which development can proceed. An attempt has been made to keep the analysis and conclusions as general as possible, although some of the conclusions made were based on a vehicle configuration (GD/C version) consisting of 12 booster

engines and 3 orbiter engines, each with a 400 000-pound thrust level. More detailed studies and changes in requirements will dictate changes and refinement to the concept.

The booster and orbiter avionics systems are considered to be completely independent; i.e., there is no signal interface between vehicles for information transfer. It is also assumed that the booster will provide all flight-control functions during the launch phase.

5.4.1 Booster

Figures 5-4, 5-5, and 5-6 are functional diagrams of the control system for the orbiter vehicle; however, they will be used as a reference in describing the booster system since many of the considerations are directly applicable to the booster. The data bus concept is depicted on Figure 5-4; but Figures 5-5 and 5-6 assume signal transfer without indicating how the data flow will be implemented.

A strapdown inertial measurement unit was selected as the basic reference system for both orbiter and booster since it is believed that this represents a worst case for the data management sizing problem as well as a realistic approach for the eventual system. The advantages and disadvantages of the strapdown versus a gimbaled platform are as follows:

- Advantages
 - The strapdown is more easily and effectively adapted for redundant operation.
 - It is somewhat more reliable since gimbal torquers, slip rings, etc., are not required.
 - It has unlimited freedom in all three axes.
 - Service and maintenance should be simpler.
 - The requirement for separate flight control rate gyros may be eliminated.
 - It requires less power, weight, volume, and cost.

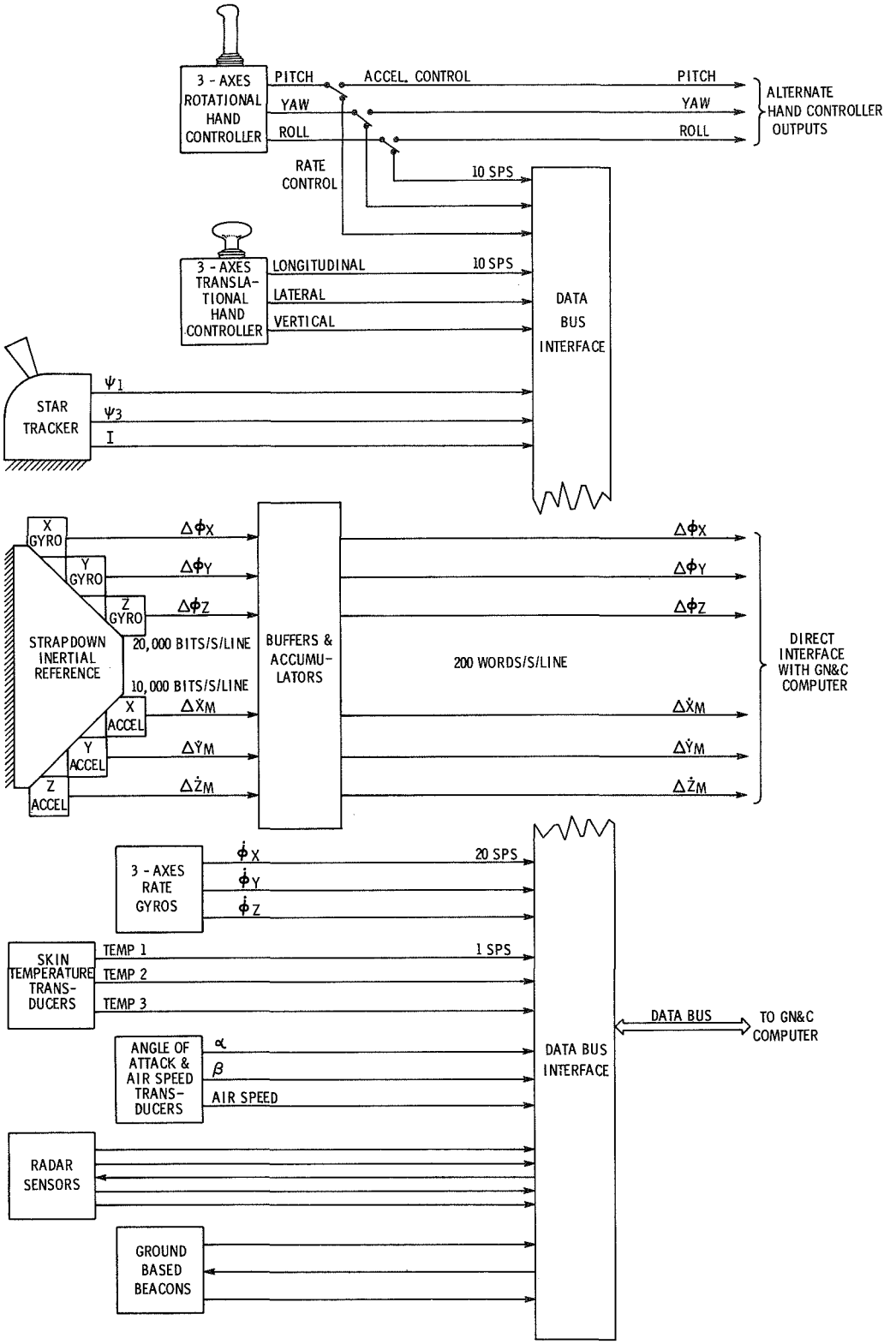


Figure 5-4. Sensors and controllers.

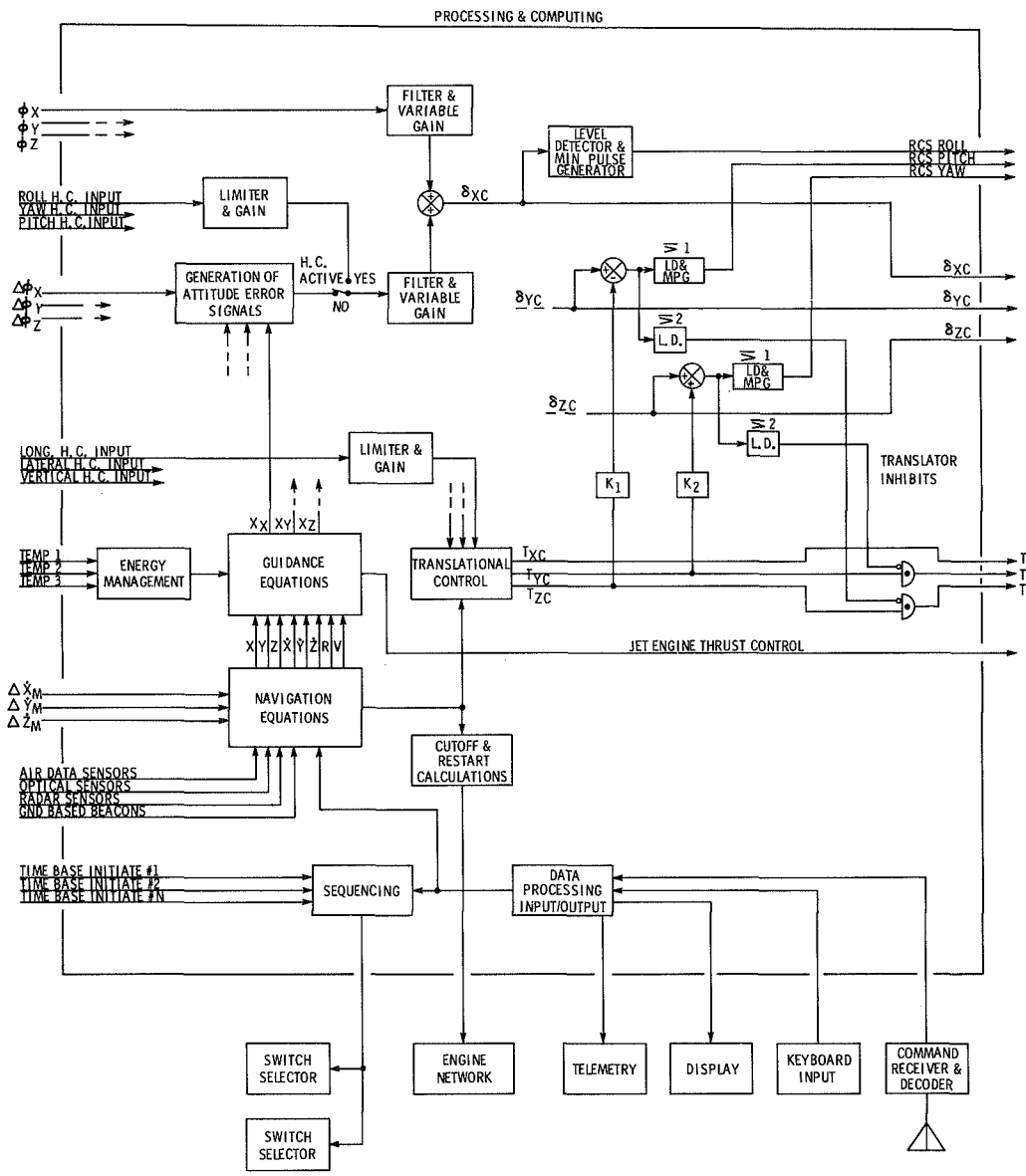


Figure 5-5. Orbiter G, N, and C flow diagram.

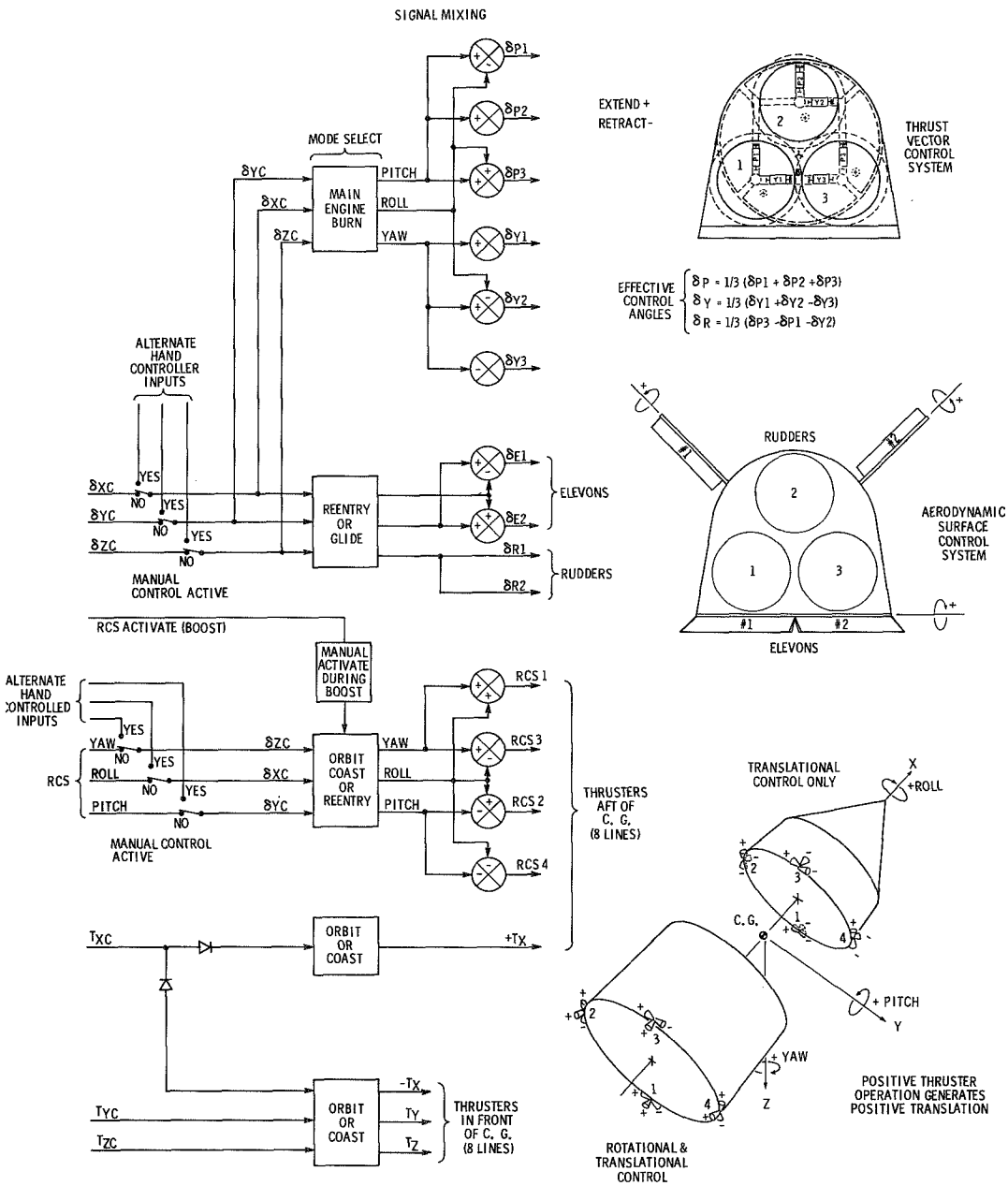


Figure 5-6. Control actuation.

- Disadvantages

 - Accuracy is not as good as gimbal systems (errors are about 2 times larger).

 - Computer memory, speed, and accuracy requirements may be increased.

 - Strapdown will require some development, whereas an acceptable gimbaled system would not.

The determining factors are based on the shuttle requirements for high reliability and its associated redundancy, ease of maintenance, low cost and, most of all, the fact that no requirement for high accuracy of the inertial system has been established.

The high data rates that are expected from the accelerometer outputs should go directly to an accumulator by hardwire, not data bus (Fig. 5-4). The output of the accumulator can go on the data bus if required; however, it is assumed that the proximity of the strapdown to the computer would make hardwires most efficient.

A separate rate gyro package is shown in Figure 5-4, but will be used only in case the rates derived from the strapdown system are not adequate to provide stability. Vehicle body bending stabilization might require rates that are measured at some location in the rear of the vehicle, as well as at points along the airframe.

Temperature transducers will be located on the skin of the vehicle to monitor heating rates during reentry. These outputs should be displayed to the pilot and possibly may be used to override the normal control commands. Such an override would change the angle of attack to lower vehicle temperature.

Angle-of-attack sensors may be available for display to the pilot, but load relief during atmospheric flight will probably be achieved by direct measurement of lift forces. This information can be taken from the body-fixed inertial measuring unit.

The processing of control sensor inputs and computation of actuation commands as shown in Figure 5-5 will probably be performed in a central digital computer system. Incremental angles or angular rates which are taken from the IMU can be converted to direction cosines to provide a reference

attitude instead of Euler angles, as shown on the diagram. Calculation of attitude error and actuation commands are carried out in a conventional manner. The requirements, if any, for self-adaptive control are unknown at this time. The signal mixing and switching between various actuation devices are not carried out in the central computer; however, this is not a clear-cut decision. The reasons for selecting remote mixing were to transfer fewer separate signals to the rear of the vehicle, to have fewer D/A conversions and, if required, to provide a place for direct input of commands to the control actuation system without going through the computer or data bus.

Sequencing signals are shown controlling switch selectors in the diagram; however, the data management concept would probably eliminate the requirement for switch selectors.

Signal mixing and mode select logic are shown in Figure 5-6 to illustrate functional operation. The details shown in the figure are not intended as recommendations for implementation. The thruster location for rotational control and aerodynamic surface control applies to the booster, but the booster requires no translational RCS thrusters. The thrust vector control scheme for one booster configuration is shown in Figure 5-7. Note that all engines are

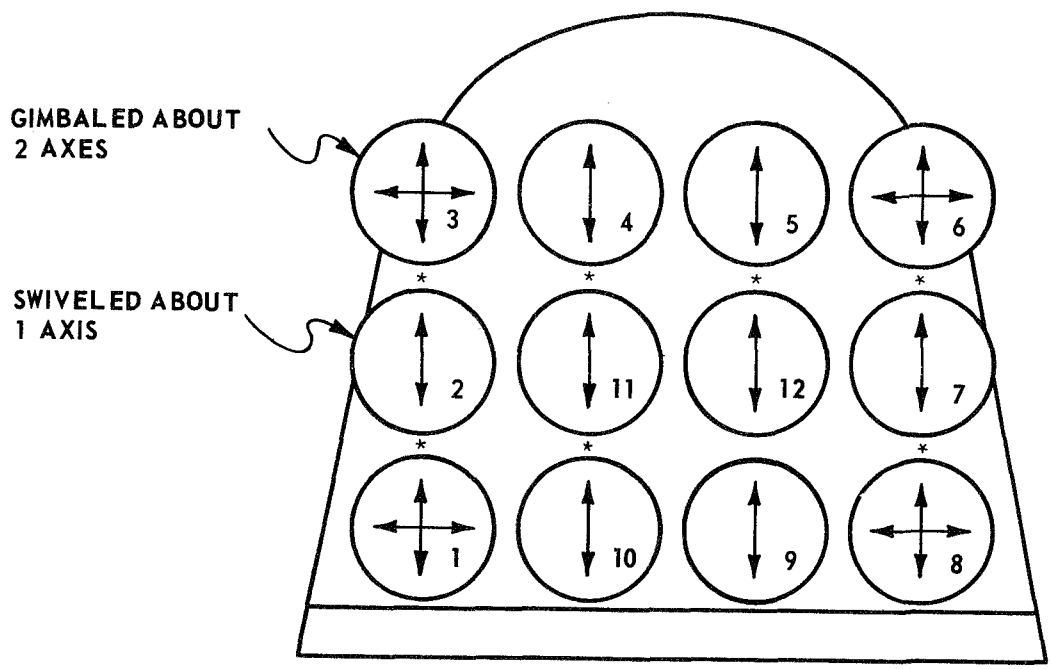


Figure 5-7. Typical thrust vector control scheme for a twelve-engine booster configuration.

swiveled (rotation about one axis) to track the cg motion, but only four are gimbaled (rotation about two axes) to provide yaw and roll control. Differential thrusting was considered as a means of vehicle control but was discarded for the following reasons:

- An engine-out produces loss of control for any configuration with fewer than 8 engines and control may be marginal for even large numbers of engines.
- Low control efficiency results from the relatively small lever arm between engines. (The distance from gimbal point to CG is 5 times the distance between engines.)
- Very complicated logic is required for precision thrust control.
- Tracking of the CG through large angles requires swiveling of engines to avoid excessive differential throttling.
- Roll control is achieved at the expense of required engine cant angles and their associated performance losses.
- Differential thrust control on the booster will not permit a lighter engine design since the same engine design must be used for control of the orbiter by gimbaling. Gimbaled control of the orbiter is mandatory to provide vehicle control after an engine-out.

5.4.2 Orbiter

Because the orbiter control system is very similar to that selected for the booster, only the differences from the previous description are discussed here.

The orbiter must have a star tracker and possibly other optical sensors to align the inertial system before reentry. During orbital coasting phases, there is also a requirement for translational control. The translation commands will be calculated in the digital computer based on hand-controller inputs or guidance commands. The translation thrusters may be integrated with rotational control thrusters. If so, translation inhibits will be required to assure priority of rotational control torques (Fig. 5-5). The manual activation of the RCS during boost could provide additional control torques in case of actuator failure or other emergency situation which might create marginal stability with the three-engine configuration (Fig. 5-6). Note that the roll command is mixed with only three of the six actuators. Roll control during aerodynamic flight is obtained by differential swiveling of the elevons.

5.5 Control System Simulation Status

Preliminary work was started on a dynamics and controls digital simulation deck early in Phase A. This was to be a simplified model which used small angle approximations and linear aerodynamic characteristics, but with enough flexibility to verify analog frozen point studies and to investigate lateral and longitudinal cross-coupling problems. It was also intended to be used as the source of data input for separate analog models. As more information became available on the shuttle, it was obvious that the large angles of attack and very nonlinear aerodynamic characteristics (Fig. 5-8) would require that the simulation be expanded.

Presently, the deck has the capability for vehicle translation in three axes about a spherical, nonrotating earth. The atmospheric model is a direct exponential function of altitude, and gravity varies as the inverse square of the distance from the center of the earth. The deck also has rotational dynamics about three axes with closed-loop vehicle controllers in all three axes. Virtually all of the aerodynamic terms have been incorporated into the deck; however, these terms are valid for only a limited flight regime and each run is limited to a particular flight interval such as the hypersonic, supersonic, etc.

About 4 months have been spent debugging this deck and a number of stable runs have been made with all translational axes active under reentry conditions. Work is now in progress to operate the simulation for other flight regimes such as atmospheric boost. A number of the aerodynamic and cross-coupling terms are set to null and have not been used actively in the program. This program runs in about real time on the CDC 3200 computer and can also be run on the Univac 1108 computer.

Analog programs in block diagram form have been prepared for simulation of both lateral and longitudinal dynamics. These programs were planned as the major element in the control and dynamics studies.

A polynominal manipulator program is available for finding roots of the characteristic equations. This is a valuable stability analysis tool which has been exercised on preliminary shuttle data. As more reliable data become available, it will be used to verify the stability of the uncontrolled vehicle and finally the vehicle with closed-loop control system for different flight intervals.

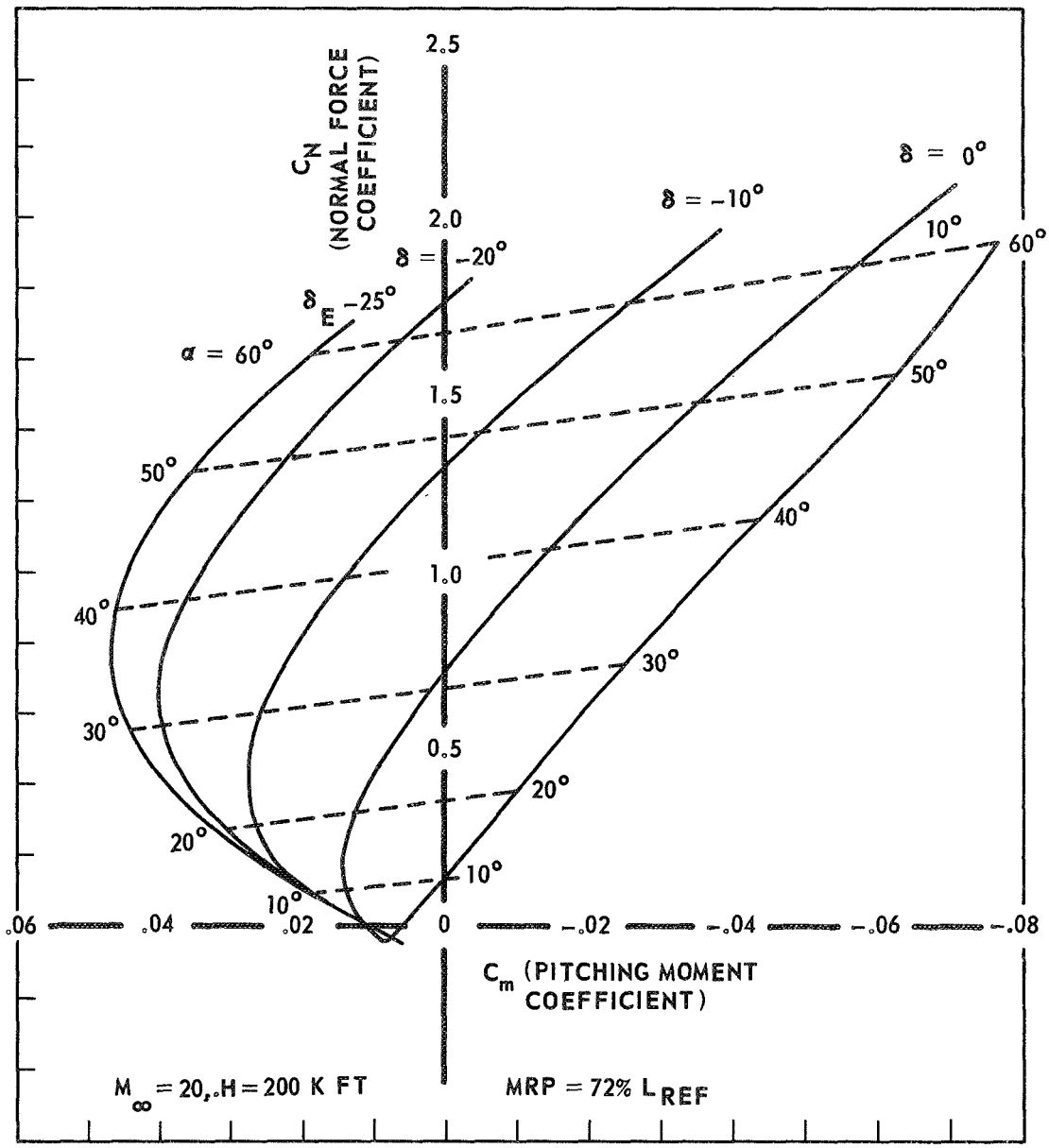


Figure 5-8. Second-stage normal force variation with pitching moment for elevon deflections of +10 degrees to -25 degrees hypersonic flow.

5.6 Navigation System Conceptual Design

5.6.1 Introduction

A preliminary conceptual design for a space shuttle navigation system has been defined to permit further development of the space shuttle systems. Navigation systems are composed essentially of two parts: the navigation sensors and the navigation computation. This description concentrates on the navigation computations required to regularly determine position and velocity during the mission. The navigation sensors are not described in detail but are treated only as providing information to the navigation computation. The navigation sensors that were considered as a function of the required mission phases are listed in Table 5-4. In this table, the symbol B indicates a preliminary selection, whereas the symbol A indicates a potential alternate or addition as well as an area for further study and trade-off analysis. A basic philosophy followed is that the position and velocity will be determined from an inertial navigation computation by integrating the outputs of the inertial measurement unit (IMU). All other sensor inputs will be used only to realign or update the inertial navigation computation.

Utilization of alignment sensor information is described for prelaunch gyrocompassing and theodolite alignment, and for orbital realignment using star tracker information. Utilization of position update information is discussed for ground based tracking, land mark tracking, VHF omnidirectional range (VOR), Tactical Air Navigation (TACAN), and the Instrument Landing System (ILS). Utilization of velocity update information is described only for ground-based tracking but will eventually include a doppler radar velocity update during the glide or cruise phase. Position and velocity update during rendezvous has not yet been covered and is not included in this description.

Primarily, only the orbiter was studied; but some of the results also apply to the booster. Realignment of the booster navigation system was not found to be required and is not planned even though booster position and velocity updates will be performed.

Several coordinate systems are required to perform the navigation function. A letter subscript on the components of a vector identify the coordinate frame. For instance X_N, Y_N, and Z_N represent the components of a vector in navigation coordinates. Similarly X_B, Y_B, and Z_B represent vector components in body coordinates. If a vector is not specifically referenced, the coordinate frame will be indicated by the appropriate letter, such as (N) for navigation coordinates or (rw) for airport runway coordinates.

TABLE 5-4. NAVIGATION SENSORS

	Prelaunch Alignment	Ascent	Orbit Maneuvers	Rend. & Docking[1]	Deorbit	Reentry	Glide	Approach & Landing
IMU	B	B	B	B	B	B	B	B
Pendulums	A							
Longitudinal Accelerometer			A	A		A		
Theodolite	A							
Star Tracker			B		B			
Sun Tracker			A		A			
Horizon Scanner			A		A			
Landmark Tracker			B		B			
Map Matcher			A		A			
Daylight Star Tracker	A							
VOR							B	
TACAN (Azimuth)							B	
ILS Localizer								B
ILS Glide Slope								B
ILS Markers								B
LORAN - C							A	
OMEGA							A	
Navigation Satellites			A	A	A		A	
AROD			A		A		A	
Scanning Fan Beam								A
TACAN (Range)							B	
DME							B	
Long-Range Rendezvous Radar				B				
Short-Range Rendezvous Radar				B				
Docking Radar/Laser				B				
Orbital Altimeter			B		B			
Doppler Radar							A	
Low Altitude Altimeter							B	B

1. not studied in this iteration
A = potential alternate or addition
B = preliminary selection

VOR — VHF Omnidirectional Range
TACAN — Tactical Air Navigation
ILS — Instrument Landing System

LORAN — Long Range Navigation
AROD — Airborne Ranging & Orbit Determination
DME — Distance Measuring Equipment

The coordinate transformation matrix (CTM) for transforming a vector from coordinate system (A) to coordinate system (B) will be indicated as CTM (A/B).

The navigation coordinate frame (N) is reestablished periodically since it drifts from its original orientation over a period of time. For realignment, a coordinate frame (N') is established which has the same orientation as the original (N) coordinate frame. The realignment is accomplished by commanding N equal to N' in the appropriate coordinate transformation.

Some of the concepts and philosophies described in this conceptual design leave room for improvement as the design continues. For example, there is no statistical processing or smoothing of navigation information. Eventually this should be incorporated to allow improvement in knowledge of the vehicle state when complete three-axis information is not available, or is available only with large uncertainties. As another example, some of the schemes described have singularities which must be eliminated.

5.6.2 Navigation System Overview

Three basic coordinate frames are utilized in Figure 5-9. The navigation coordinate frame (N) is aligned along the north pole and the vernal equinox. The body coordinate frame (B) is aligned along the vehicle roll, yaw, and pitch axes. The third coordinate frame assumes that a strapdown IMU will be selected; however, only the name of the coordinate frame will change if a stabilized platform IMU is chosen. The strapdown inertial coordinates (SDI) are kept in the strapdown computer by continuously generating the coordinate transformation matrix relating body coordinates (B) to this coordinate system. They are allowed to drift without ever being reoriented. For this conceptual design all alignments and updates are performed in the main computation routine and not in the strapdown computation, although performing these functions within the strapdown computation is certainly an attractive alternate approach.

There are three types of inputs to the system: the inertial measurements, alignment data, and position and velocity update data. Measured velocity from the IMU is rotated into navigation coordinates, compensated for gravity, integrated and presented to the guidance computation, which then determines the desired vehicle attitude and thrust level. Alignment information from the sensors determines a CTM relating the correct navigation coordinates (N') to body coordinates (B). Combining this with the information from the strapdown computation defining body attitude, CTM (B/SDI), an alignment matrix CTM (SDI/N') is established for realignment.

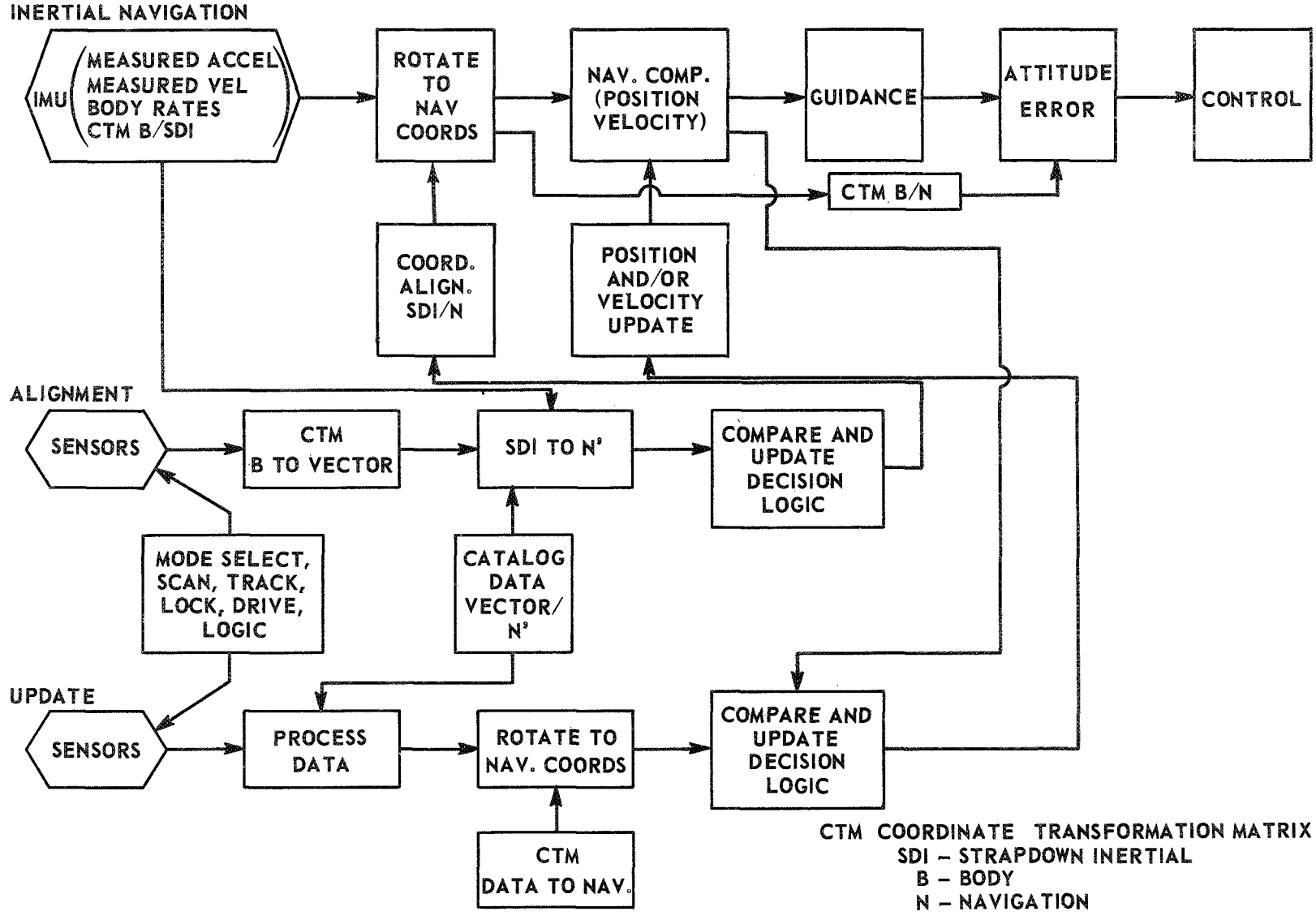


Figure 5-9. Navigation system overview, general block diagram.

Position and velocity update sensor information is processed and rotated to (N) coordinates for possible use in the navigation update computation.

5.6.3 Inertial Navigation and Update Scheme

Measured velocity from the IMU is rotated from SDI to N coordinates, compensated for gravity, and integrated to find the navigation state. Rectangular integration is shown in Figure 5-10 because it is simple and illustrates the position update scheme. In actuality, various higher order as well as alternate or special purpose integration schemes should be considered.

The first step in realigning the navigation system is to normalize the CTM (SDI/N') received from the realignment sensors. The comparison and update decision logic shown in Figure 5-10 uses direction cosines; however, this logic could be replaced by the equivalent test for other coordinate transformation techniques.

A temporary memory is provided to store the navigation state for periodic times (t_1). All position and velocity update information must either be read at, or projected to, this time. Velocity update information is incorporated in the gravity compensation equation as a velocity bias equal to the difference at time t_1 of computed velocity components and measured velocity components. Position update is accomplished by adjusting the constant of integration, at t_1, by an amount equal to the difference of computed position and position measured by the position update sensors.

5.6.4 Prelaunch Alignment

Figure 5-11 allows for either theodolite or earth gyrocompass alignment. Both alignment schemes depend on the strapdown IMU to provide the gravity vector components. This makes both schemes sensitive to accelerometer bias errors.

The theodolite alignment is straightforward using programmed knowledge of the rotating earth to establish a CTM (L/N') for local (L) to navigation coordinates. This is combined with the CTM (B/L) from the theodolite and CTM (B/SDI) from the strapdown computer to obtain the required CTM (SDI/N') for alignment.

Gyrocompassing during the prelaunch alignment phase will be performed entirely within the computer by using information from the onboard strapdown instruments (Fig. 5-10). Past studies of several techniques for computational gyrocompassing indicate that gyro drift is the major source of error, whereas location of the vertical is the second most important error source.

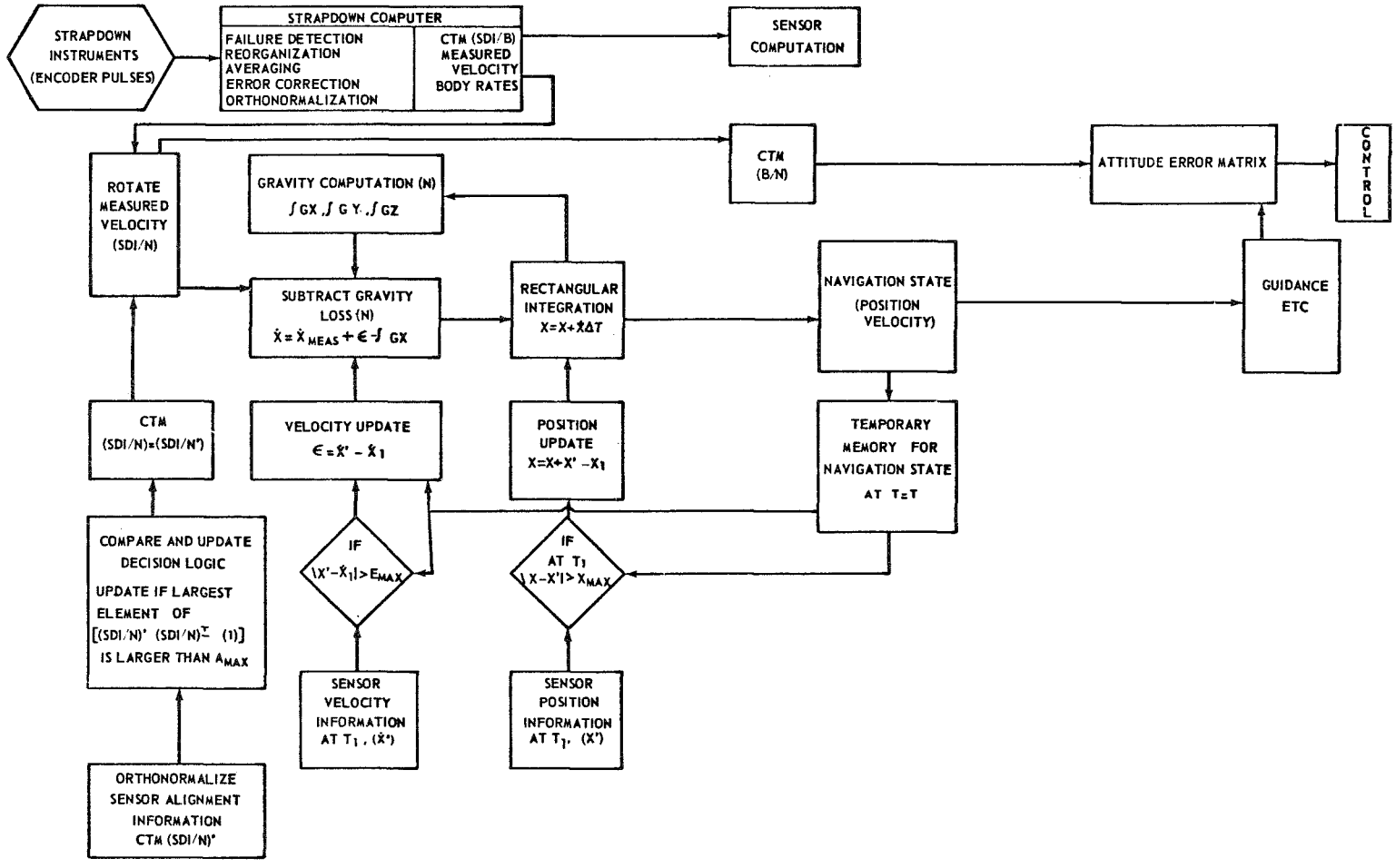


Figure 5-10. Inertial navigation and update scheme.

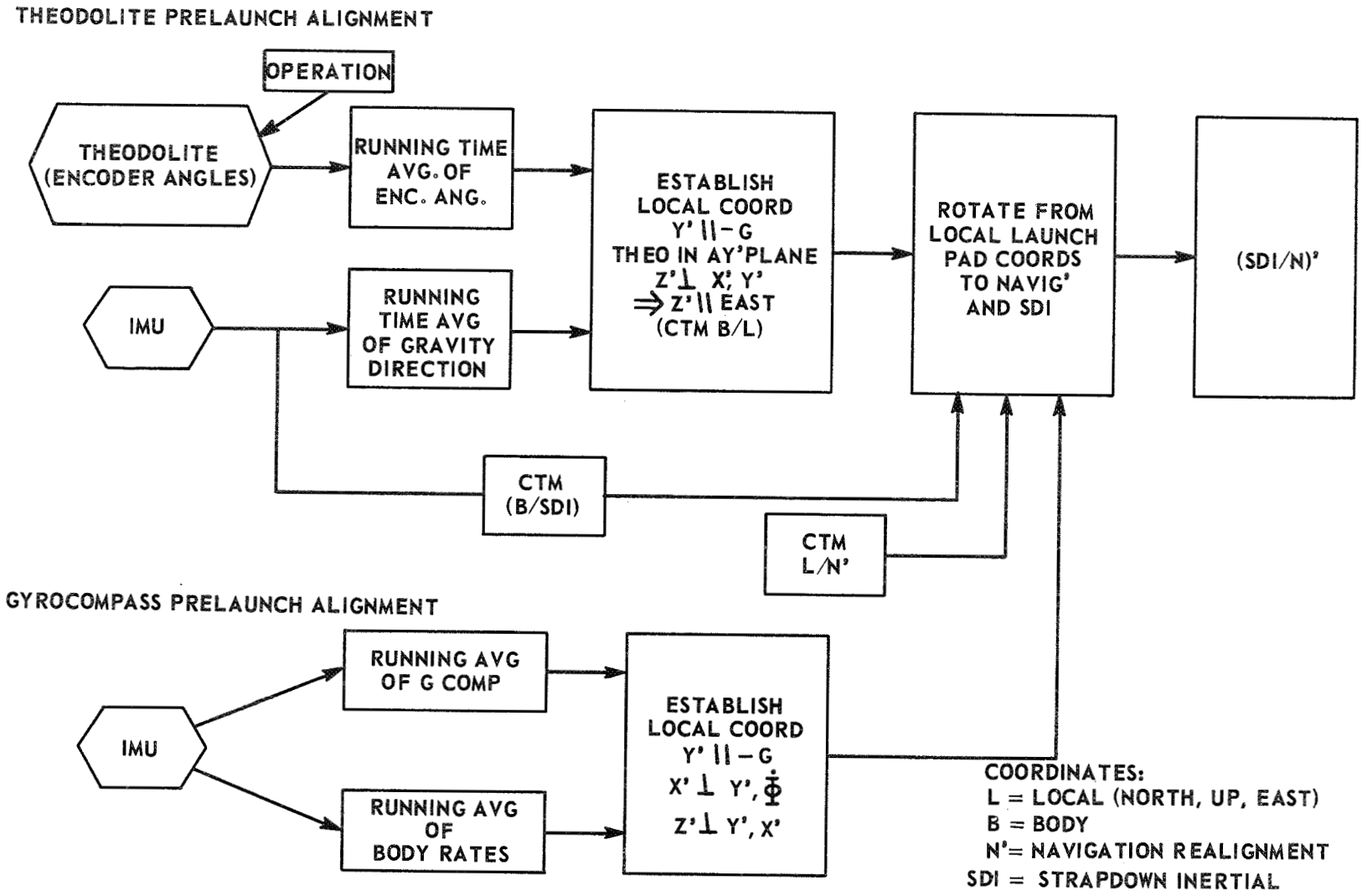


Figure 5-11. Prelaunch alignment.

5.6.5 Star Tracker Orbital Realignment

Simultaneous observations of the direction to two different known stars are required in body coordinates with the scheme shown in Figure 5-12 to establish the coordinate transformation matrix CTM (B/N'). This is combined with the IMU output CTM (SDI/B) to obtain the required CTM (SDI/N'). Star catalog data are required.

In the event that the reference is completely lost, other sensors will be required to effect a crude alignment. Such sensors might be a horizon scanner or sun tracker. The scheme shown in Figure 5-12 does not take this into account.

5.6.6 Orbital Update

Two methods of obtaining update information are ground-based tracking and vehicle-based landmark tracking (Fig. 5-13). Using ground based tracking information, the orbital parameters are projected to time t_1, converted to position and velocity and presented to the inertial navigator. Using landmark tracking, angles are measured simultaneously to two landmarks and combined with orbital altitude to establish spacecraft position. This position is ambiguous since it does not establish on which side of the line connecting the landmarks the vehicle is located. A third landmark can resolve the ambiguity if it is not colinear with the first two.

5.6.7 VOR Two-Station Position Update

A VOR station provides vehicle azimuth or bearing relative to that station. With the two equations as shown in Figure 5-14 one can solve for X_1 and Z_1, the vehicle location in a local horizontal plane with respect to VOR Station 1 for a vehicle not on a line connecting the two stations. This, with radar altimeter data and local terrain elevation data, locates the vehicle in earth-fixed (W) coordinates. If the vehicle is on a line connecting these two stations, an input from a third station not on this line is required. Elevations of the VOR stations and of the terrain may not be required as onboard data for adequate navigation. An error analysis will determine the sensitivity of the navigation system to the elevation information.

5.6.8 VORTAC Position Update

Since a VORTAC station (a station consisting of both TACAN and VOR) provides both bearing and range relative to a given station, data from a radar altimeter will complete the information required for position determination, as illustrated by Figure 5-15. To properly update earth-fixed

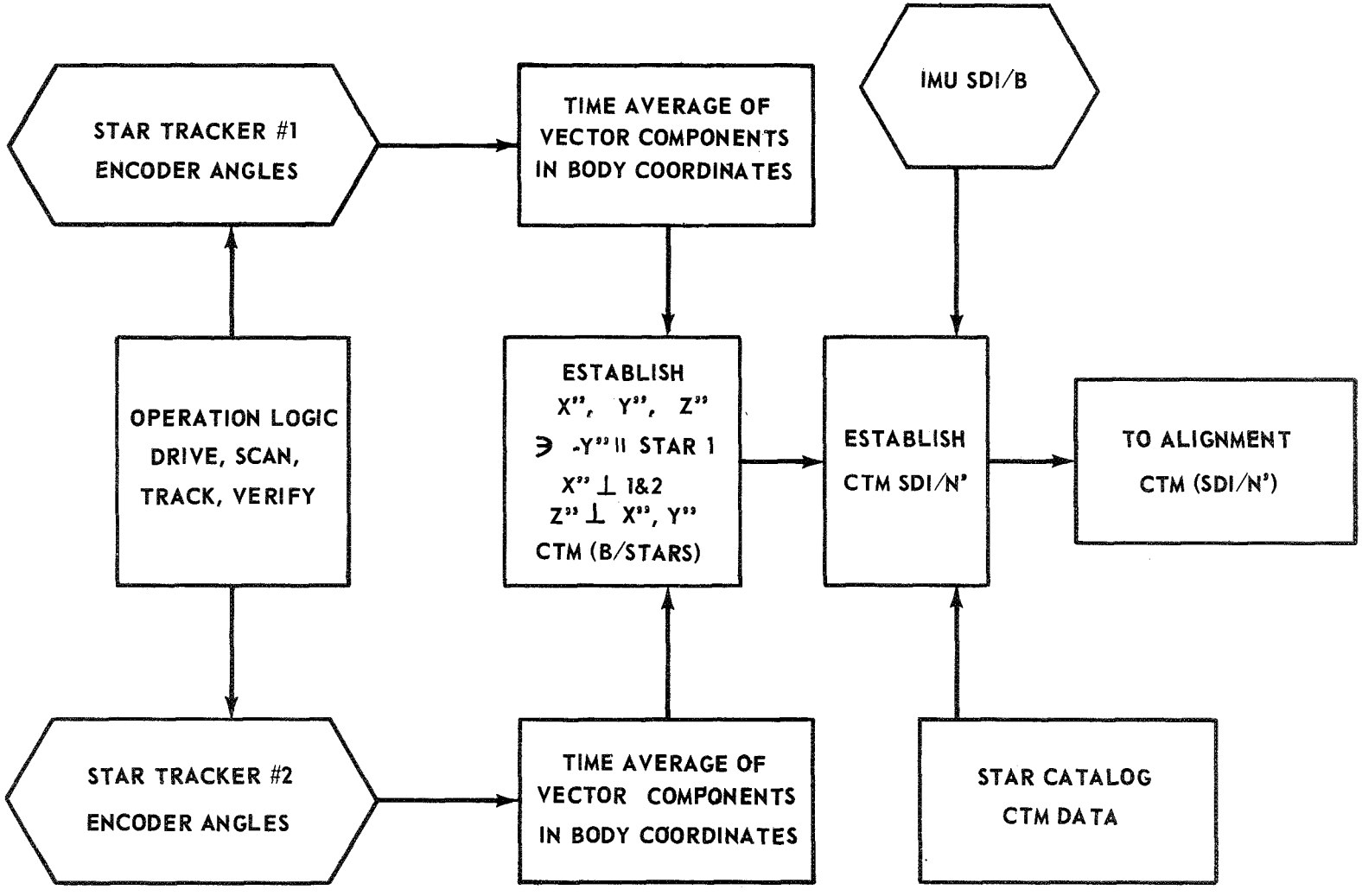


Figure 5-12. Star tracker orbital realignment.

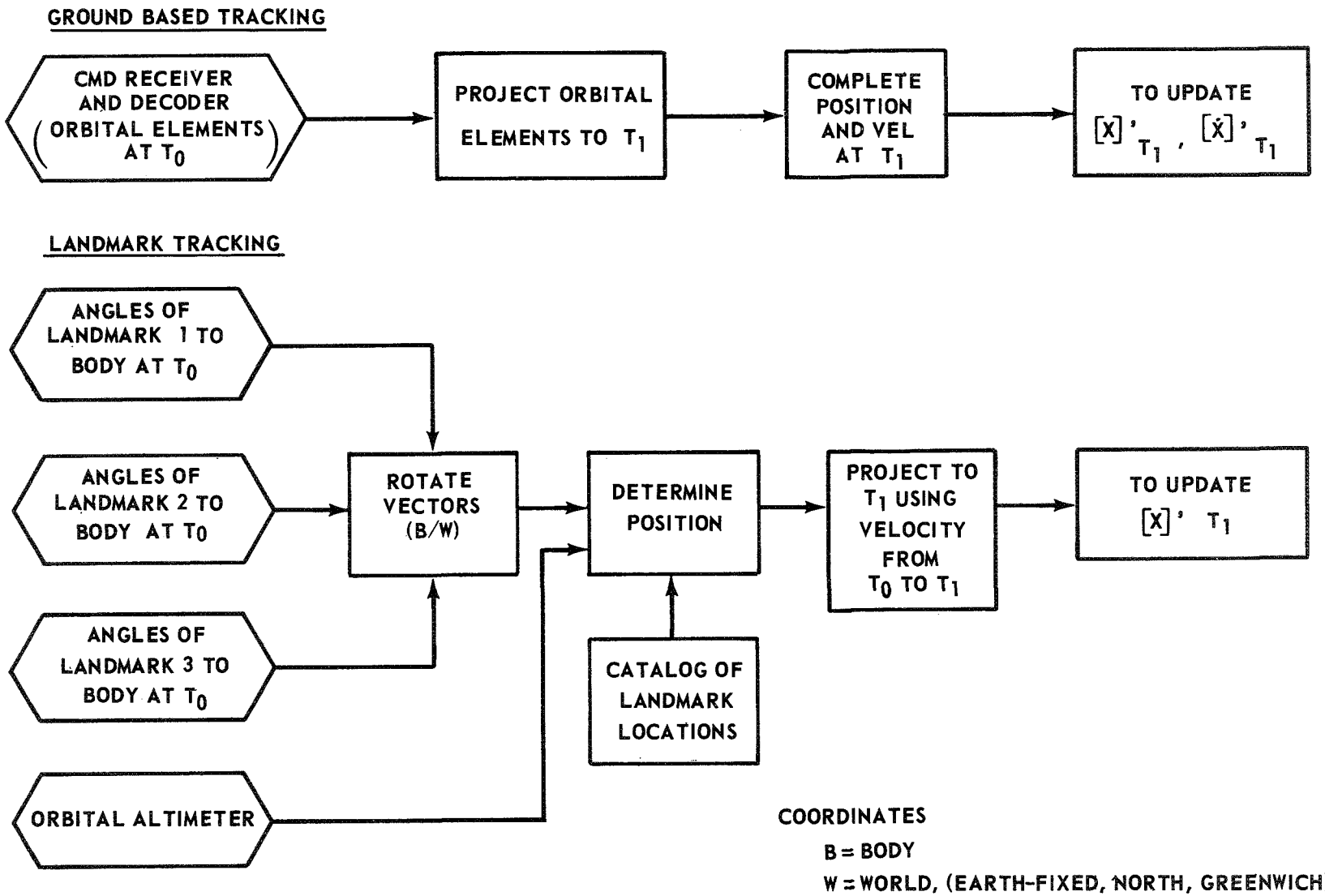


Figure 5-13. Orbital update.

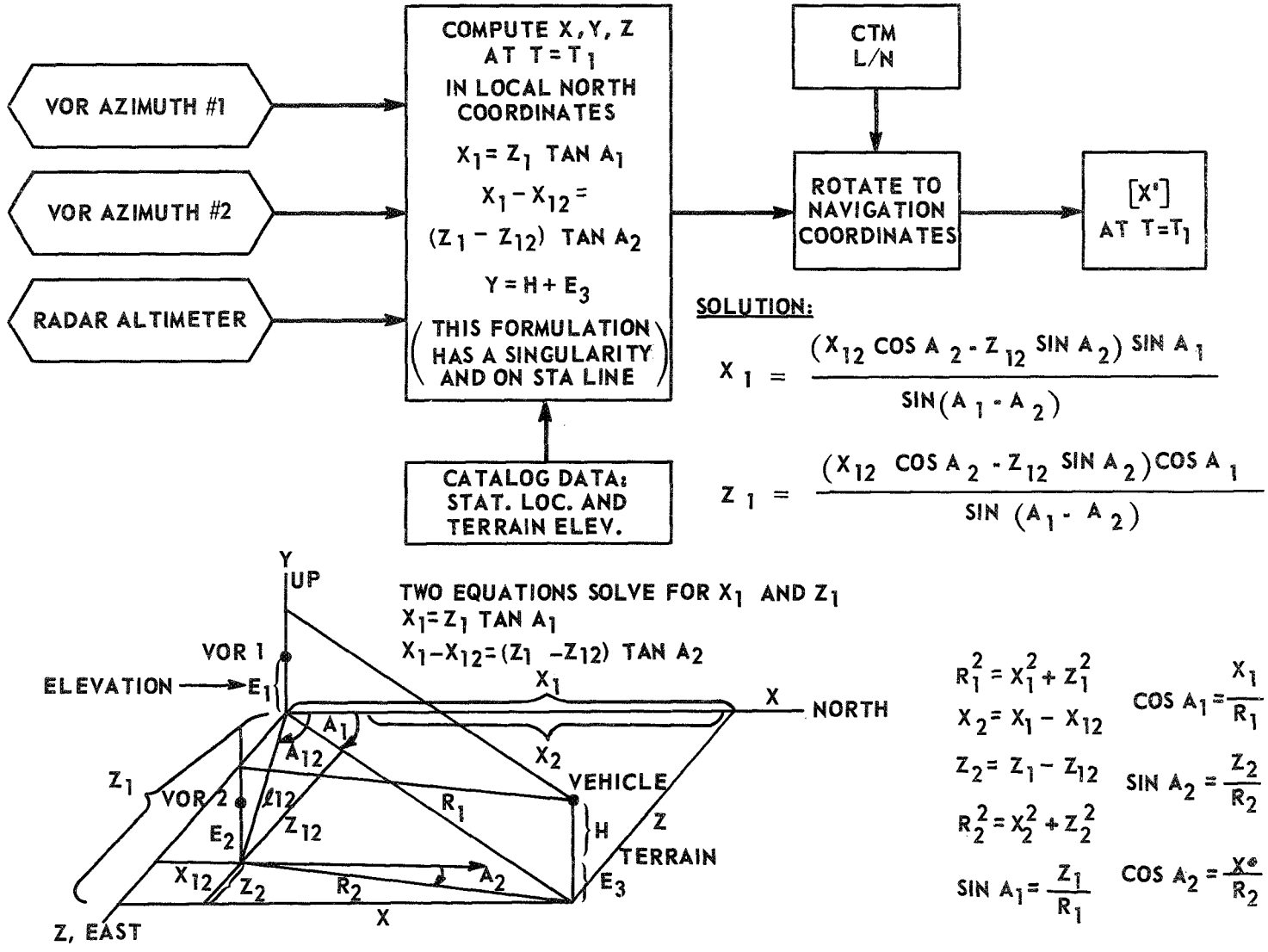


Figure 5-14. VOR 2 Station position update.

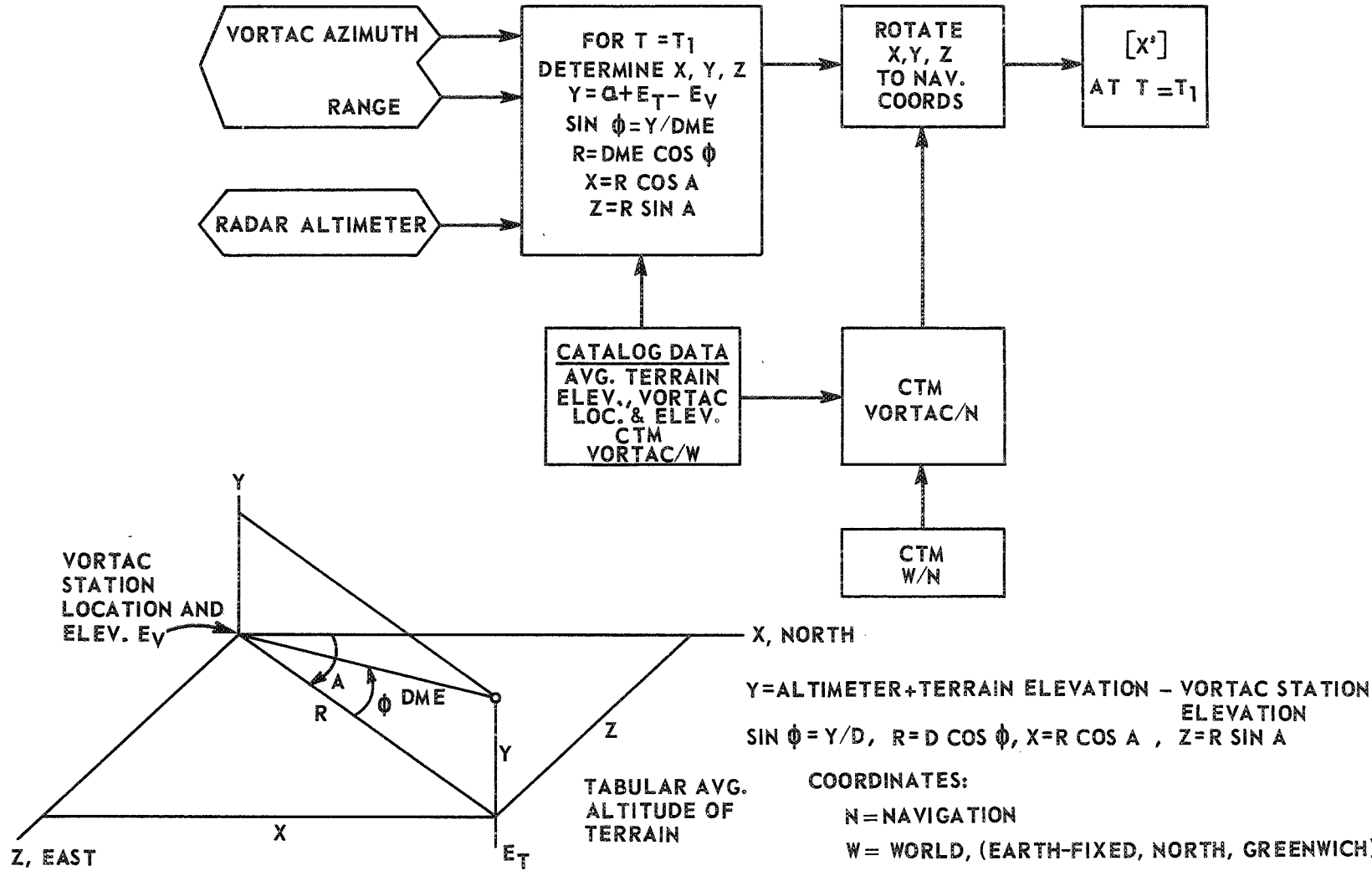


Figure 5-15. VORTAC position update.

coordinates, the average sea level elevation, the elevation of the ground beneath the vehicle, and other VORTAC station elevation may be required. This data may be included in a coarse form in the computer catalog of data. Barometric altitude may prove to be more desirable than radar altimeter compensated with cataloged ground elevations.

5.6.9 ILS Position Update

Because of multipath problems with the glide slope and localizer beams (Fig. 5-16), some type of smoothing will be required for the update system. A radar altimeter is utilized to do some of the smoothing of the glide slope irregularities in the critical vertical channel. Use of this altimeter will require accurate knowledge of the terrain beneath the vehicle.

In Figure 5-16 the glide slope deviations, localizer deviations, and DME indications determine the position which is converted to navigation coordinates. The Y channel is compared against the radar altimeter for verification and passed on to the navigation update.

5.7 Reentry Navigation Hardware Error Analysis

A first-order simulation was conducted of the performance of an inertial platform during the reentry phase. The simulation is first order in the sense that gravity was assumed constant, entry angles were assumed shallow, path angles were assumed small, the trajectory was constrained to a single plane, and the atmosphere was assumed to be logarithmic with altitude. Experience has shown that these assumptions give reasonable initial results for preliminary error analysis investigations.

Reference trajectory guidance was used in the simulation with L/D modulated by angle of attack as a function of deviations from nominal altitude, altitude rate, range, and range rate. The trajectory attitude maneuvers were identical to those required by the reference trajectory to damp the phugoid oscillations and to prevent skip-out.

The reference trajectory has a constant L/D throughout reentry. As illustrated by Figure 5-17, it begins at 122 kilometers (400,000 feet), an entry angle of 1.5 degrees, and near circular orbital velocity. A constant 30-degree roll angle about the velocity vector was required down to the first skip point where the roll angle was then varied to maintain constant altitude until equilibrium glide was established. Zero roll angle was assumed thereafter. The roll angle causes lateral translations, but these were not considered in this analysis. Reentry was assumed to terminate at 30.5 kilometers (100 000 feet) and the cruise phase was assumed to begin at this point with a position update from some ground-based radio navigation aid.

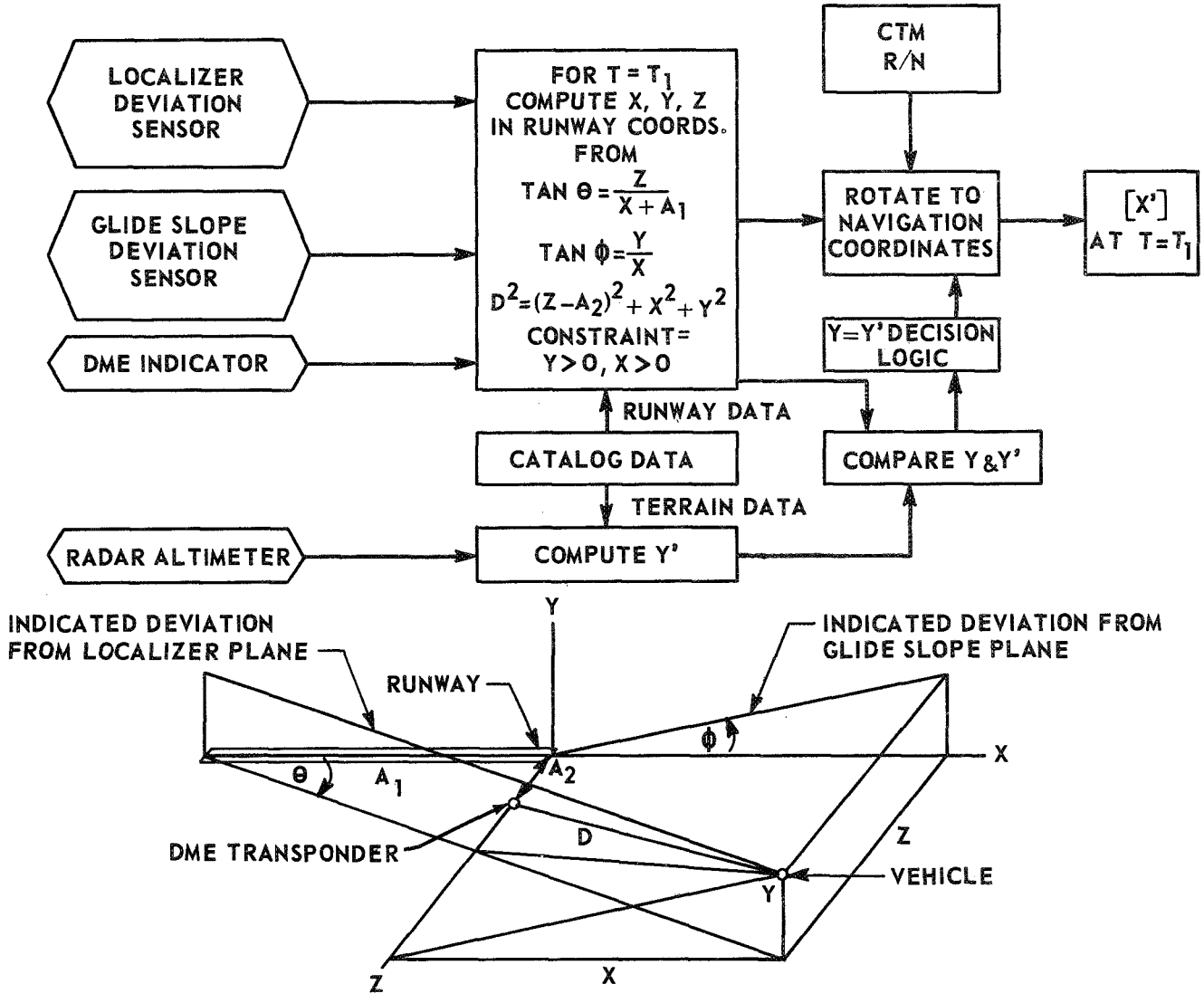
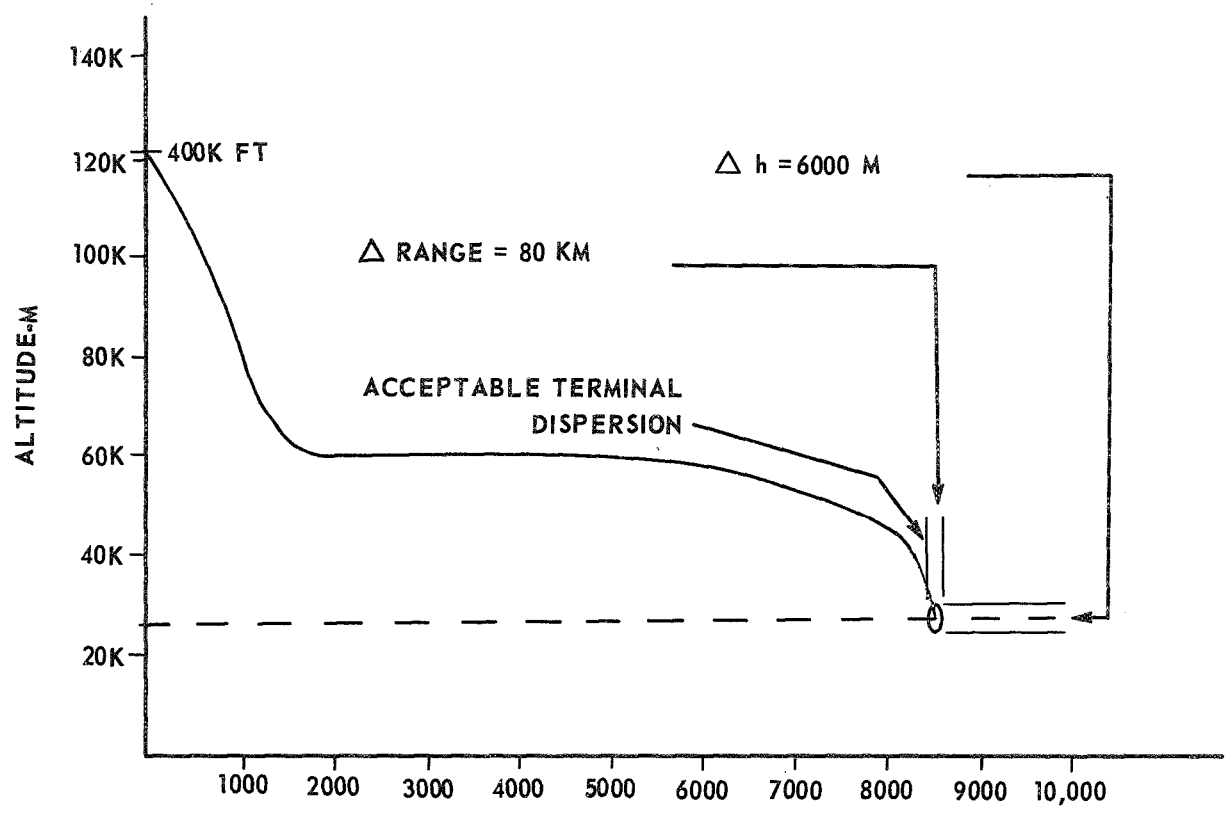


Figure 5-16. ILS position update.



I.C.
ENTRY ANGLE = 1.5°
VELOCITY = 7457 M/S
$(L/D)_{MAX}$ = 2.0

$\frac{C_D A}{M}$ = 0.128

INITIAL ROLL ANGLE = 30°

Figure 5-17. Reentry profile.

The error analysis was written for platform type errors rather than for strapdown type errors for three reasons: (1) the platform errors were simple to incorporate into a digital program; (2) it was believed at the time that high accuracy would be required for the reentry phase; and (3) a strapdown system was already under study by contractors. Error sources considered for the inplane analysis were accelerometer bias and scale factor along the X and Y axes, platform misalignment and drift about the Z axis, and a mass unbalance on the input axis of the Z gyro. The other errors either contributed to cross-range errors, which were not considered here, or were known from past experience to be negligible. All platform drifts which occurred prior to 122 kilometers (400 000 feet) were considered as misalignments so that the computer program began with zero accumulated drift angle.

The guidance law was closed through the navigation hardware which resulted in two types of errors that must be considered. The navigation estimate error is the difference between where the vehicle is and where it is computed to be using a perfect navigation scheme, but with imperfect navigation sensors. The terminal dispersion error is the difference between the reference trajectory terminal conditions and guided (or misguided) trajectory terminal conditions. For vacuum flight, the two types of errors are essentially equivalent; but, for reentry, the two types are quite different. This difference is due primarily to the logarithmic nature of the atmosphere which couples range rate to altitude. Any guidance scheme which involves range and range rate will tend to guide the vehicle to the reference atmospheric density and drag. This automatically constrains vehicle altitude even though the knowledge of inertially derived altitude degrades. The results of the two analyses in Figure 5-18 show the two types of errors with the indicated error regions and indicate that a price is paid for the altitude constraint. There is a degree of loss of control in range even though the navigation system can measure the range adequately. A guidance scheme can probably be chosen which will minimize this deviation.

Tables 5-5 and 5-6 are the performance results of two sets of different quality inertial navigation equipment. The first set is of equipment that is representative of the performance of operationally demonstrated equipment (such as 0.1 degree/hour drift). Star tracker realignment of the platform was assumed to have been made about 4 minutes prior to entering the reentry phase (122 kilometers altitude). The second case is representative of the quality of performance of readily available equipment (such as 0.25 degree/hour drift), with star tracker alignment about 12 minutes prior to entering the reentry phase. Both cases assumed a ground-based tracking update of position and velocity just prior to reentry. For comparison purposes, an

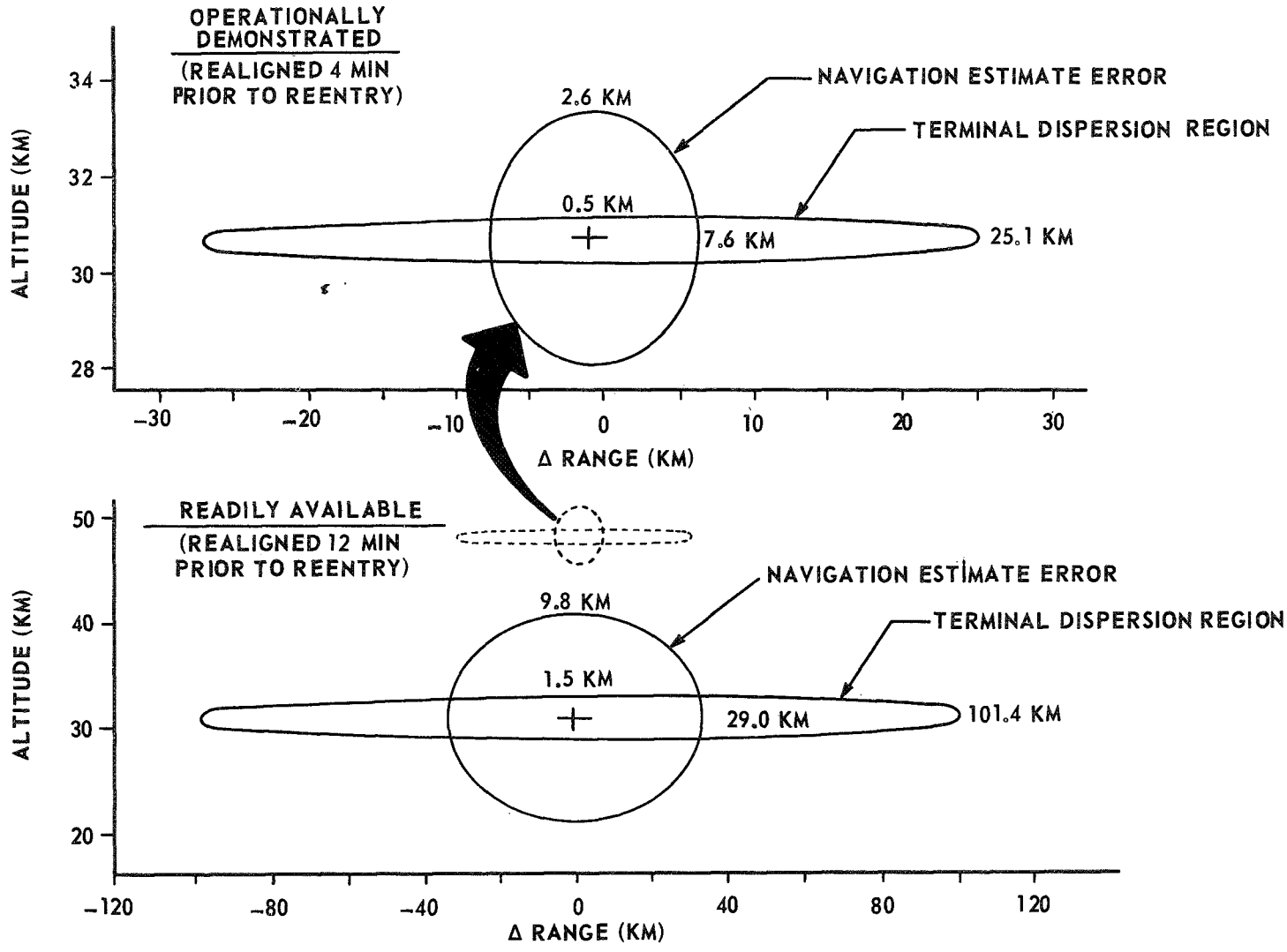


Figure 5-18. Reentry terminal dispersions for pure inertial navigation.

TABLE 5-5. REENTRY GUIDANCE HARDWARE ERROR ANALYSIS

State-of-the-Art Instrumentation	Terminal Dispersions				Navigation Estimate Error			
	Δh (m)	Δ Range (km)	$\Delta \dot{h}$ (m/sec)	$\Delta \dot{S}$ (m/sec)	$E_{\dot{h}}$ (m)	E Range (km)	$E_{\dot{h}}$ (m/sec)	$E_{\dot{S}}$ (m/sec)
Accelerometers								
(10^{-5}g) X-Bias	- 16.38	- 0.84	-0.05	- 0.89	93.88	.129	0.13	0.12
(10^{-5}g) Y-Bias	- 12.35	- 1.06	-0.04	- 0.70	135.24	-0.10	0.12	- 0.13
(10^{-5}g) X-Scale Factor	- 93.33	- 0.31	-0.03	- 0.49	41.88	0.03	0.08	0.04
(10^{-5}g) Y-Scale Factor	- 12.08	- 1.09	-0.04	- 0.70	137.10	-0.08	0.11	- 0.11
Platform								
(1 min.) Misalignments	296.61	17.55	0.99	16.24	-1668.83	-5.22	-1.93	- 5.47
(0.1°/hr) Drifts	491.17	18.68	1.77	25.71	-2017.88	-5.44	-3.38	- 8.34
(0.1°/hr) G-Dep. Drifts	-150.49	- 3.27	-0.60	- 7.26	429.74	1.11	1.63	2.59
RSS	592.92	25.92	2.07	31.30	2649.23	7.57	4.02	10.41

TABLE 5-6. REENTRY GUIDANCE HARDWARE ERROR ANALYSIS

Operationally Proved Instrumentation	Terminal Dispersion				Terminal Navigation Estimate Error			
	Δh (m)	Δ Range (km)	$\Delta \dot{h}$ (m/sec)	$\Delta \dot{S}$ (m/sec)	$E \dot{h}$ (m)	E Range (km)	$E \dot{h}$ (m/sec)	$E \dot{S}$ (m/sec)
Accelerometers								
(10^{-4}g) X-Bias	- 163.8	- 8.4	-0.55	- 8.85	938.8	12.88	1.28	1.16
(10^{-4}g) Y-Bias	- 123.5	-10.6	-0.40	- 7.02	1352.4	- 0.97	1.16	- 1.28
(10^{-4}g/g) X-Scale Factor	- 933.3	- 3.1	-0.33	- 4.89	418.8	0.32	0.76	0.40
(10^{-4}g/g) Y-Scale Factor	- 120.8	-10.9	-0.37	- 7.02	1371.0	- 0.81	1.10	- 1.07
Platform								
(3 min.) misalignment	889.7	52.6	2.93	48.50	-6471.5	-15.47	- 7.69	-16.39
(0.25°/hr) drifts	1228.5	46.7	4.42	64.35	-5044.7	-13.84	- 8.49	-21.30
(0.25°/hr/g) G-Dep. Drifts	- 376.1	- 8.1	-1.49	-18.30	1074.5	2.74	2.62	6.50
RSS	1583.2	74.1	4.57	83.87	6471.5	20.93	11.90	27.76

acceptable terminal dispersion was defined as ±3 kilometers of altitude and ±40 kilometers in range from the end point of the reference reentry trajectory. This represents about 25 percent of the total glide capability in range.

To draw conclusions from the tables, one must examine the largest error contributors as well as the RSS of all contributors. The RSS results indicate that the operationally-demonstrated instruments fall well within the acceptable region, while the readily available instruments fall well outside. Since the major error contributors are misalignment and drift, an improvement of these sources of errors in the readily available instruments could bring the performance of this instrumentation within the acceptable limits.

The small allowable misalignment in both cases indicates that star tracker alignment would be advisable in a more or less continuous mode from the end of the deorbit burn until the stars are no longer visible or until the reentry maneuvers begin. Since strapdown instruments will be available with a quality comparable to that of operationally demonstrated instruments, these results do not rule out the potential use of strapdown instruments.

This error analysis does not include the propagation of initial condition errors. For this analysis the position and velocity at the beginning of reentry were assumed to have been precisely determined by a means such as perfect orbital navigation. If a horizon scanner of 0.25-degree accuracy had been used for position update, the initial condition errors would be about 25 kilometers in range and cross range. Experience has shown that this initial 25 kilometers error would probably cause a final range error of 25 kilometers or greater. For this reason, the horizon scanner is not considered in this design as an acceptable navigation sensor for position update or for precise alignment.

6.0 Communication Subsystem

6.1 Introduction

The communication subsystem provides for two-way flow of information between the shuttle, space station, other ancillary vehicles, and ground stations. The information will normally consist of voice, data, and commands. The subsystem also provides television and intercommunication between crew members.

6.2 Requirements

In keeping with the philosophy that the shuttle is autonomous, or without external control, the communication subsystem provides a minimum of external communication facilities consistent with safety and the practices of present-day commercial and military aircraft. The Federal Aviation Administration (FAA) has minimum communication standards for aircraft operating within the continental area and other controlled areas [12]. Certain mission phases may require a navigation up-date; for example, the landing phase requires coupling to ground-based RF navigation aids.

During the development and qualification tests of the vehicle, voice, data, and command capability to the ground will be required for all mission phases. These are required so that ground personnel may monitor and analyze vehicle performance to confirm the design or to determine required modifications.

The operational vehicle will be capable of performing, onboard, all the functions normally performed on ground during a space mission. For this reason, the vehicle will require very little external communications. The communications which are required are called out in the following paragraphs.

The booster stage of the shuttle, if unmanned, will require data and command capability from the ground at all times. These are required for ground control to fly the booster back. If the booster is manned, voice communication will be required between the orbiter and booster during separation. Voice will be required between the orbiter and the space station during rendezvous, docking, and cargo loading and unloading. During the glide or cruise and landing phases, the orbiter will require voice and navigation aids as required by the FAA. Communication will not be required during the reentry phase when communications are normally "blacked out" due to plasma generated by reentry heat.

6.3 Communication Subsystem

Figure 6-1 is a functional block diagram of the communication subsystem. Included in the communication subsystem are an S-band transponder for communication with the Manned Space Flight Network (MSFN), and ultrahigh frequency (UHF) and very high frequency (VHF) tranceivers for short-range ground and docking communication. The signal processor controls the signal flow between various system elements in response to data bus commands. The path of transmitted and received information may be selected manually or automatically through a preprogrammed set of instructions. Included in Figure 6-1 are elements whose function is in the guidance and control area. These functions will be discussed under their respective heading; however, the electronic requirements will be included in the communication area. These elements include radar, VHF omni range (VOR) and Tactical Air Navigation (TACAN), or VORTAC, Air Traffic Control Transponder (ATC), and Instrument Landing System equipment (ILS).

6.3.1 S-Band Transponder

The S-band transponder consisting of a transmitter and receiver is functionally equivalent to transponders presently operational on Apollo missions. The receiver is capable of simultaneously demodulating one voice channel, one up data link at 1 kilobit per second, and a pseudo random noise range code (PRN). The transmitter is capable of simultaneously transmitting one voice channel, one 52 kilobit per second telemetry channel, and retransmitting the PRN range code. An alternate mode of operation is the transmission of wide-band signals such as television. Figure 6-2 is a spectrum of the up-link signal to the S-band transponder. The carrier signal is at 2101 MHz. The voice channel is phase modulated onto a 30-kHz subcarrier, the subcarrier is then phase modulated onto the S-band carrier. The up data are frequency modulated onto a 70-kHz subcarrier which is then phase modulated onto the S-band carrier. The PRN range code is phase modulated onto the S-band carrier at baseband.

Figure 6-3 is a spectrum of the down-link signal of the S-band transponder. The down-link voice channel is phase modulated onto a 1.25-MHz subcarrier which is then phase modulated onto the S-band carrier. The down-link data (telemetry) are biphase modulated onto a 1.024-MHz subcarrier which is then phase modulated onto the S-band carrier. Note that the 1.024- and 1.24-MHz subcarriers are placed near a null in the PRN spectrum to help prevent interference between the signals. The spectrum of the alternate mode of operation of the transmitter is also shown in Figure 6-3. In this mode, television or other wideband signals may replace the PRN modulation.

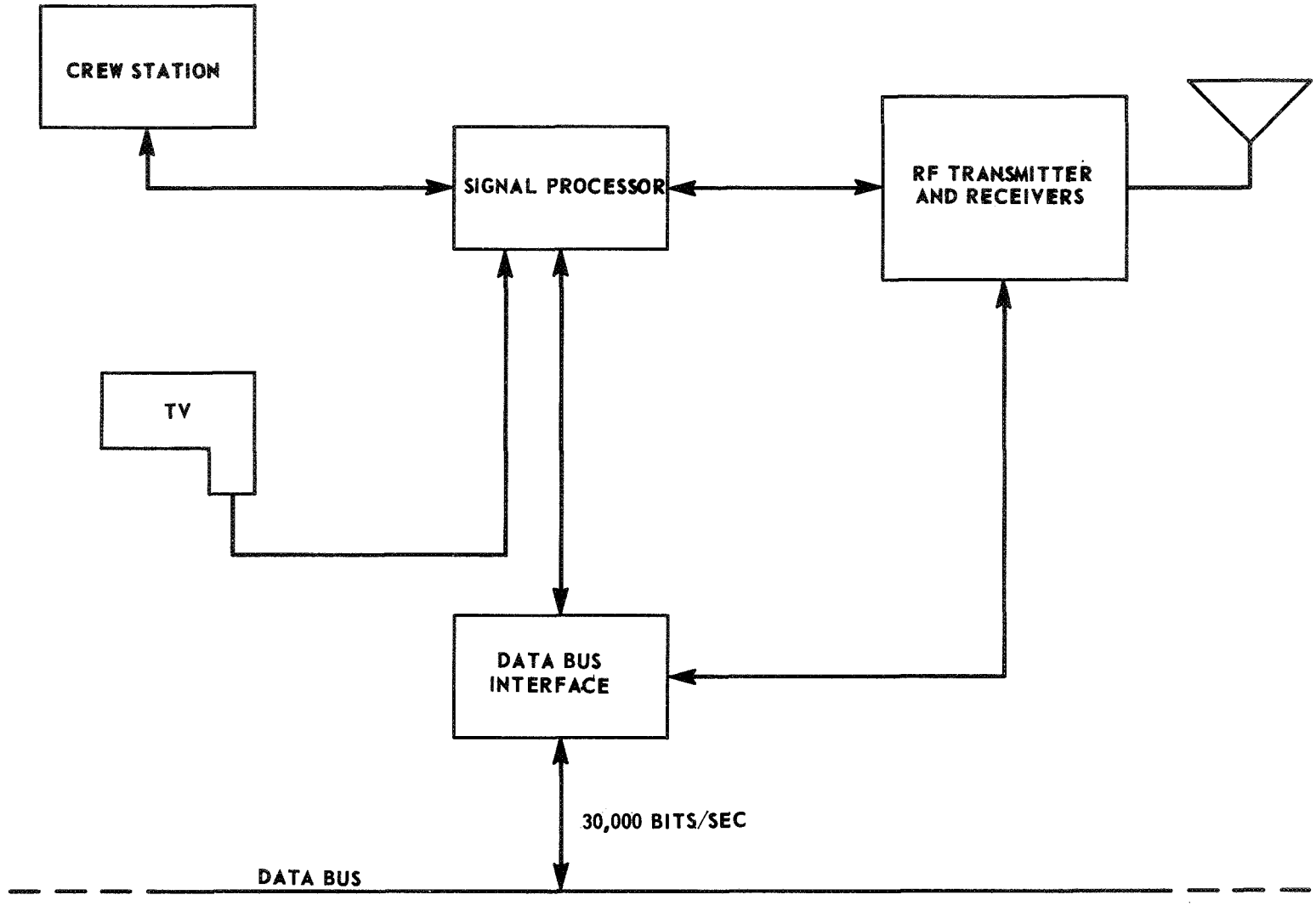


Figure 6-1. Communication system functional block diagram.

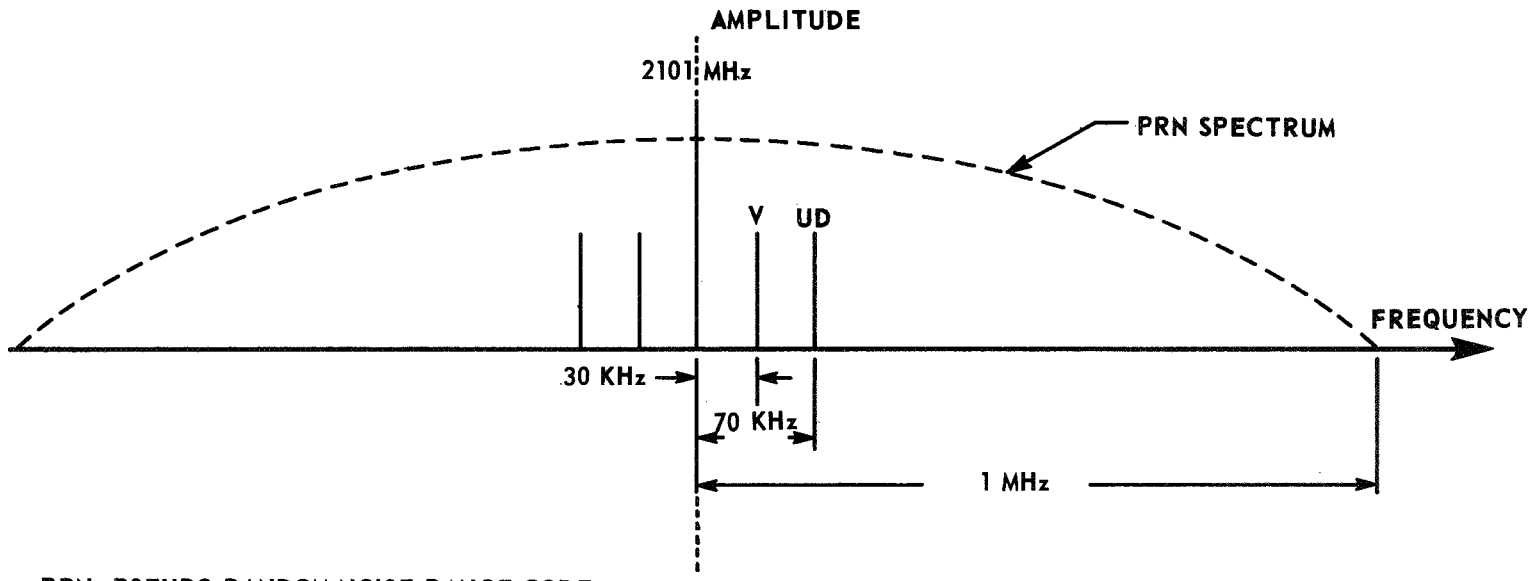


Figure 6-2. S-band up-link spectrum.

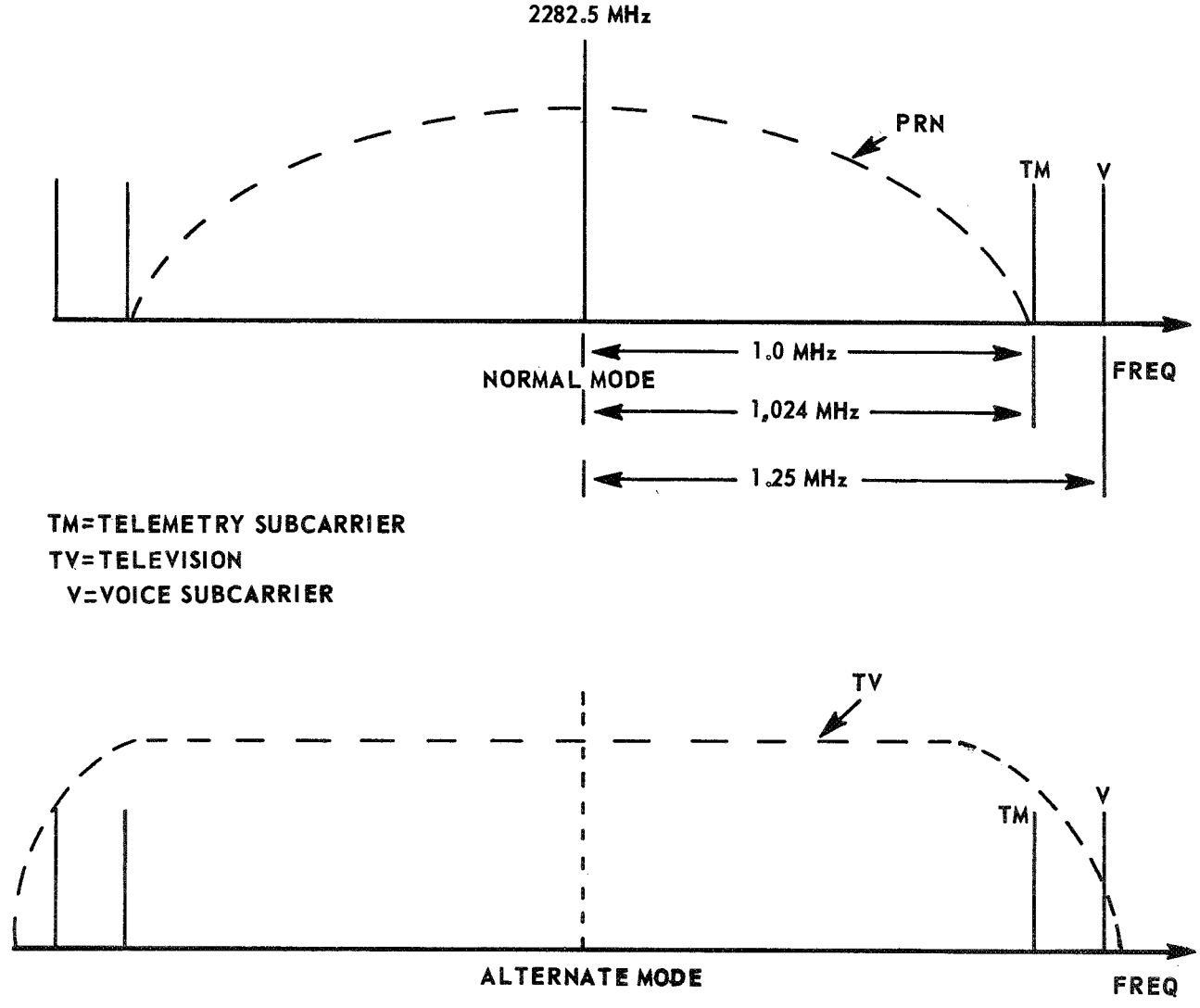


Figure 6-3. S-band down-link spectrum.

The S-band link is dependent upon the MSFN being operational. GSFC, the controlling agency of the MSFN, is presently planning to keep the Mila, Bermuda, Madrid, Goldstone, and Honeysuckle stations active during the 1970-1980 period[1].

6.3.2 VHF and UHF

The VHF and UHF transceivers are primarily for short-range voice communication with the ground and voice and data communication with the space station. The voice link will be used with the space station during docking, cargo transfer, and extravehicular activity. The link with the ground will primarily be as required by FAA during ferrying, landing, and take-off phases. These units will have a multichannel capability to operate with all radio facilities likely to be encountered during the mission. The maximum range to the space station for voice transmission is 3700 km (2000 n. mi.) and 370 km (200 n. mi.) for 10-kilobits per second data transmission.

6.3.3 Television

Television will be used by the crew primarily as a visual aid to monitor vehicle configuration, to view equipment in inaccessible areas, and to aid in landing, docking, and cargo handling. This will require several TV cameras depending upon the vehicle configuration. Since the video signal of the TV is wideband (4.5 MHz), it is not recommended that it be carried on the data bus. The video signal should be hardwired as shown in Figure 6-4 to a central video switch. The crew members may then select the camera they wish to monitor via data bus commands. If desirable, cameras may be gimbal mounted and controlled by the crew members via the data bus.

6.3.4 Signal Processor

The signal processor has three functions; it acts as a switching terminal to transfer or switch information from the source to the user, it controls the amplitude of the signals, and it modulates and demodulates signals to and from the RF communication equipment. The signal processor has no decision capability. It is controlled entirely by data bus inputs.

6.3.5 Intercommunication

The intercommunication function consists of voice communication between crew members and from crew members to external terminals, such as the space station. Since few voice channels will be required, it is recommended that voice signals not be carried on the data bus. Reasonable fidelity

1. Telephone conversation between T. A. Barr (MSFC) W. Varson and T. Roberts (GSFC).

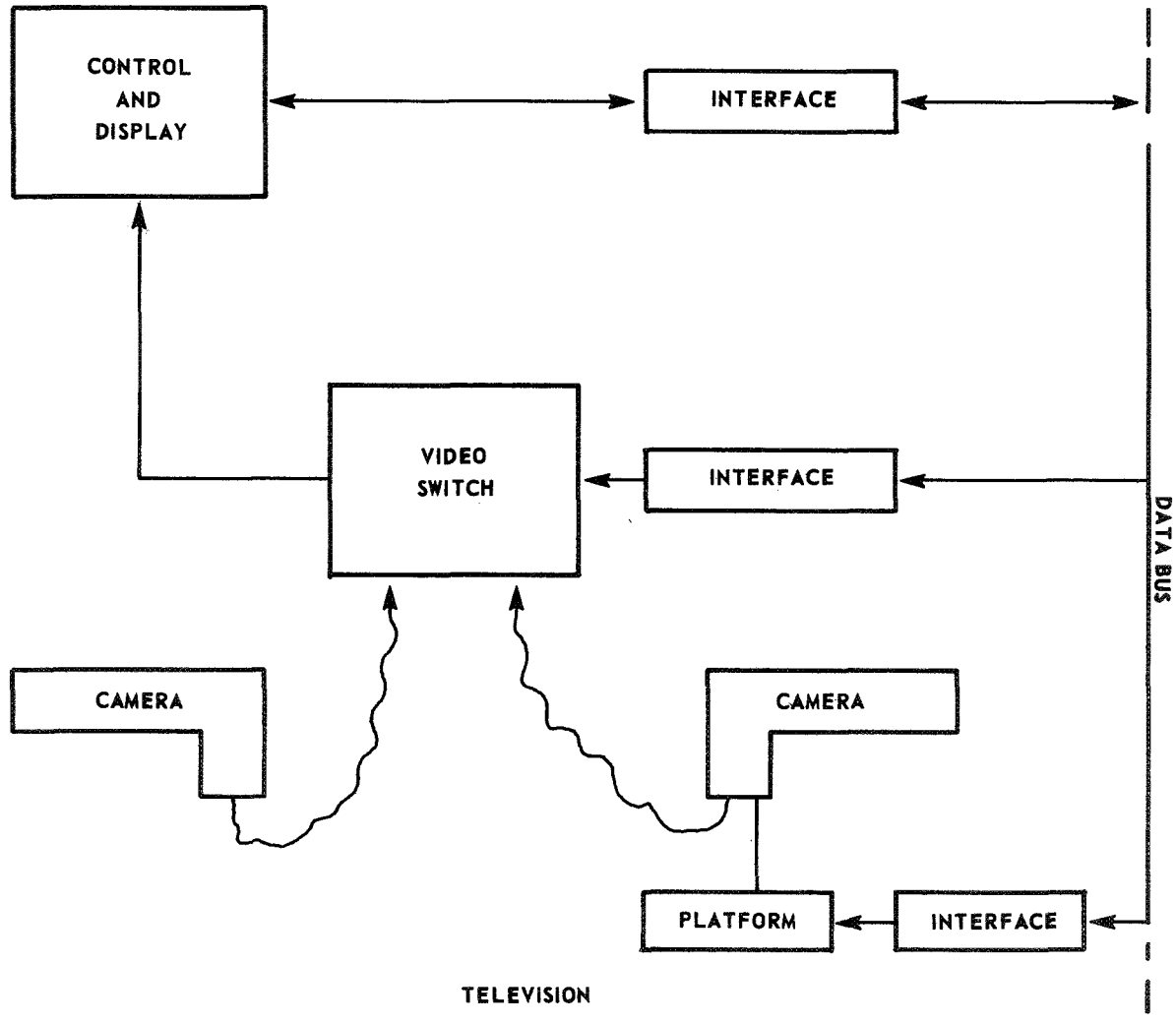


Figure 6-4. Hardwired video signal.

with digitized voice signals would require approximately 150 kilobits per second (10 bit word, 3 kHz, 5 samples per cycle) per voice channel, assuming no data compression techniques. The data bus as presently envisioned has capacity for digitized voice. Should the audio hardwire weight become excessive in the final design, then a voice signal can be carried on the data bus.

6.4 RF Navigation Aids

Commercial and military airports in the continental U. S., and major overseas terminals, have common navigation and landing aid frequencies. These aids use radio frequency techniques to provide bearing and distance to the airports and airway facilities, and glide slope and azimuth information during landing.

6.4.1 Instrument Landing System (ILS)

ILS is a radio frequency (RF) navigation aid that provides lateral and vertical path deviation data and range markers during the landing phase of the mission.

The lateral (localizer) error signal is provided by a transmitter with two identical antenna patterns as shown in Figure 6-5. The right pattern signal is modulated by 150 Hz and the left by 90 Hz. Normally these signals are demodulated by a receiver in the vehicle and are used to drive an indicator right or left depending on vehicle location relative to the antenna pattern or runway. The RF transmission is on one of 20 channels from 108 to 112 MHz. Vertical error signals (glide slope) are provided in a manner similar to the lateral signals. In this case, an upper antenna beam is modulated by 90 Hz and a lower beam by 150 Hz as shown in Figure 6-6. These demodulated signals are used to drive an indicator up or down depending on the vehicle position relative to the glide slope. The RF transmission is on 1 of 20 channels from 325 to 329 MHz.

These 20 channels are shared by the airports; consequently one channel is assigned to several airports. The sharing airports are geographically separated such that there is no interference, and the possibility of improper station selection is reduced through station identification codes.

Markers, or along course signals, are transmitted vertically by a beacon in a fan beam at 75 MHz. Three marker beacons are provided along the approach path with one near the end of the runway, one at approximately 3500 feet out, and the third about 5 miles out.

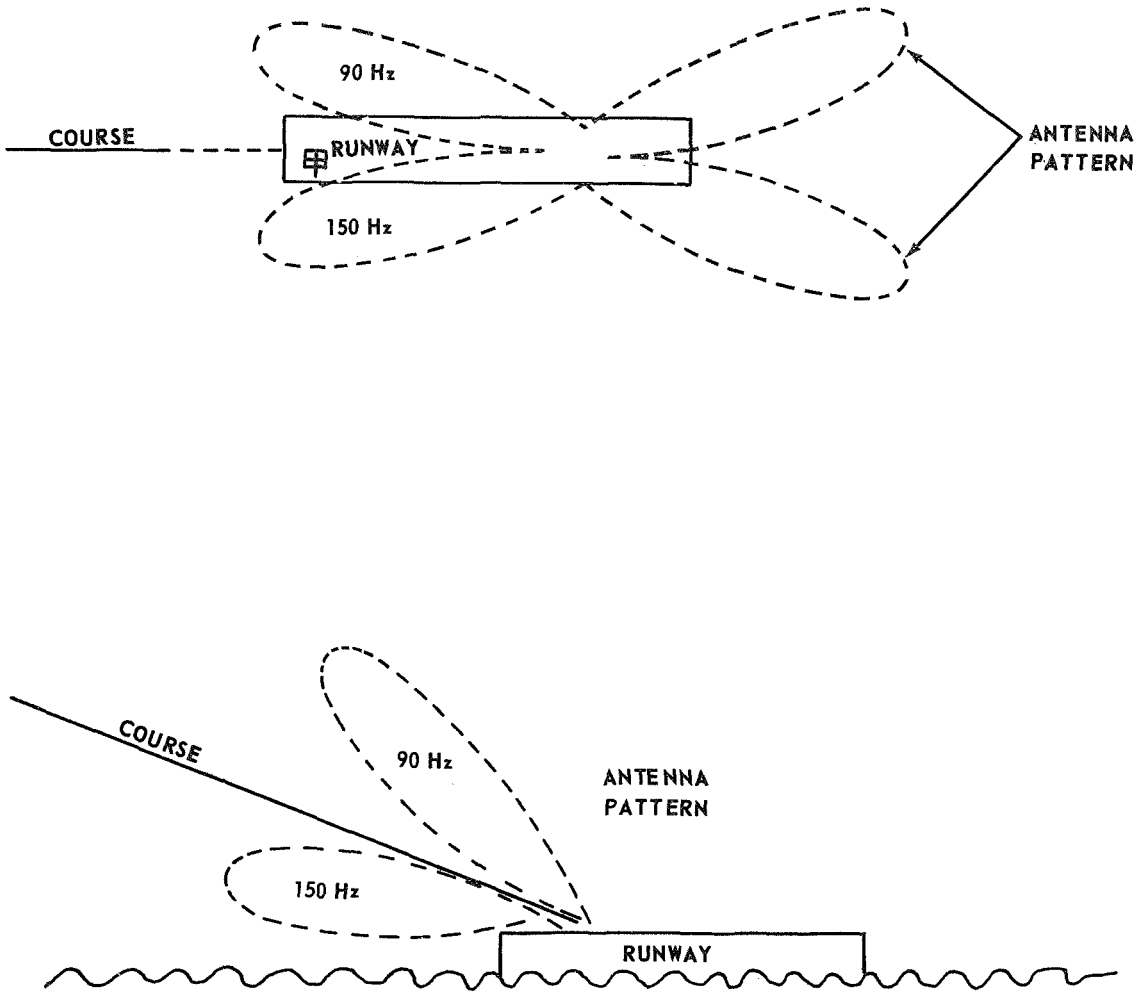


Figure 6-6. ILS vertical guidance.

The primary changes to the conventional equipment for shuttle use are associated with the data bus interface. The changes involve vertical and lateral error signal pick-off, station identification, selection of operating frequency, and automatic checkout.

6.4.2 VORTAC

VORTAC consists of co-located VOR and TACAN stations. TACAN is a radio navigation system producing range and bearing information from a

ground station. The system is active in that the aircraft system interrogates the ground system to get a response. The elapsed time between the interrogation and response is the basis for range determination.

Bearing data are obtained through rotation of the ground station antenna as shown in Figure 6-7. The VOR antenna pattern is a cardioid which rotates 30 times per second, generating 30-Hz amplitude modulation (AM) in the airborne receiver. A 30-Hz reference is also radiated which is frequency modulated (FM) ±480 Hz on a 9960 Hz subcarrier. The airborne receiver reads bearing as a function of phase angle between the AM and FM 30-Hz modulations. Figure 6-8 shows the TACAN antenna pattern which is a cardioid pattern superimposed on a 9-lobed pattern, generating 15-Hz coarse bearing data and 135-Hz fine bearing data. Reference signals are transmitted by coded pulse bursts.

Like ILS, shuttle VORTAC equipment will require changes to provide the data bus interfaces to pick off the navigation information and to control frequency.

The TACAN system operates on 126 two-way channels spaced at 1 MHz intervals between 962 and 1213 MHz. Each ground station can accommodate up to 100 aircraft at a time.

VOR operates on 20 channels between 108 and 112 MHz and 60 channels from 112 to 118 MHz.

6.4.3 Radar

Radar will be used for several different purposes on the shuttle. Most of these uses will not require simultaneous operation. Thus, to optimize weight, volume, and power, combinations of multipurpose radars and single purpose radars should be considered. The various uses or modes of operation for radar are listed in the following sections.

6.4.3.1 Rendezvous Radar

The rendezvous radar should be capable of tracking passive and active targets. The passive or skin tracking mode will be used for locating satellites and other noncooperative targets. The effective target area will be 10 square meters and the maximum range 183 km (100 n.mi.). The active mode will be used for rendezvous with the space station or another space vehicle. In this mode, the target (space station) has a transponder that

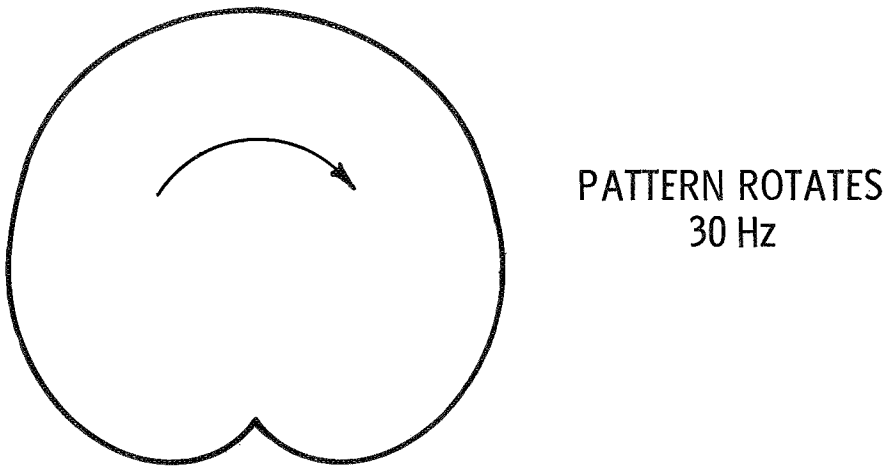


Figure 6-7. VOR antenna pattern.

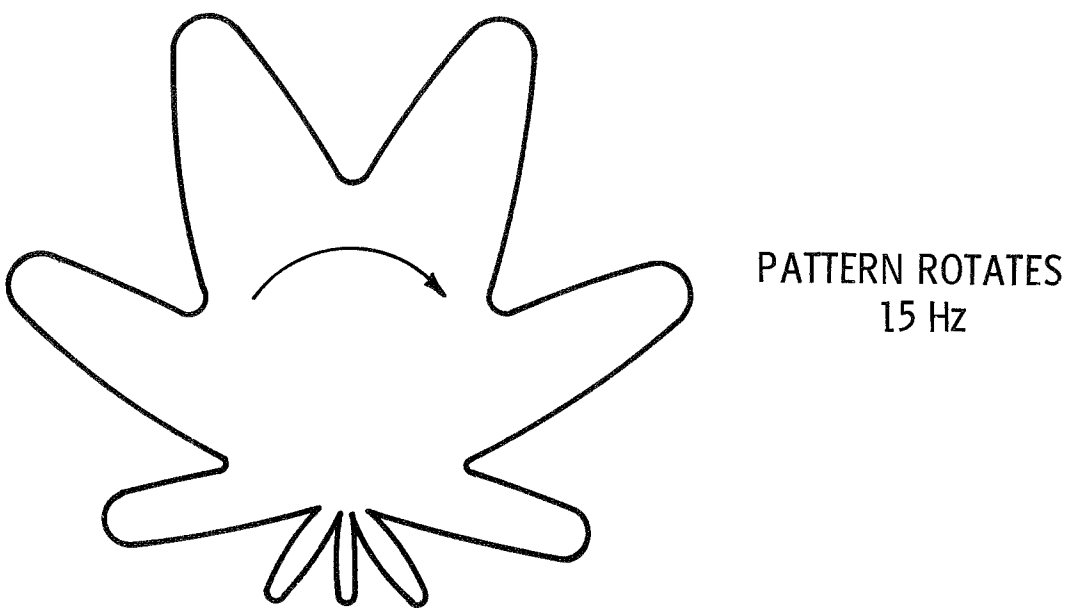


Figure 6-8. TACAN antenna pattern.

responds to an interrogation by the shuttle. Range and range rate will be determined by the interrogation and response signals. The maximum range will be 4300 km.

6. 4. 3. 2 Radar Altimeter

The radar altimeter will be used to determine shuttle altitude to an altitude of 300 n. mi. The radar will be of a pulse type in which altitude is

determined by the elapsed time between transmitted and reflected pulses. The pulse repetition frequency and pulse width should be variable such that the radar is capable of operating to the lowest practical altitude during the landing phase.

6. 4. 3. 3 Doppler Radar

A doppler radar is included to provide aircraft velocity with reference to its antenna frame coordinates. This velocity will be used, after appropriate coordinate transformations, to update the inertial navigation system.

6. 5 Antennas

With the communication system as presently envisioned there will be very little sharing of antennas. The antennas are a special problem due to the reusability requirement and subjection to a high temperature environment during reentry. The problem is further complicated by the fact that some of the antenna systems require an omnidirectional pattern which will be heavily influenced by vehicle shape and antenna location.

Further detailed study is highly recommended for the antennas since it appears that they may require technological advancement to operate satisfactorily.

6. 6 Signal Margin

Signal margin calculations for the various links show that the transmitter power and the antenna gain required for a good signal are readily obtained using present-day technology and hardware.

Tables 6-1 and 6-2 are margin calculations for the S-band link. All desired signal to noise ratios shown in the tables are as good as or better than those considered acceptable for Apollo missions. These calculations show that omni antennas and a 20-watt transmitter are sufficient for earth orbit communications with the MSFN ground stations.

Short-range link calculations are not included since these links are presently operational in commercial and military aircraft.

TABLE 6-1. MARGIN CALCULATIONS FOR THE S-BAND LINK

MSFN TO SHUTTLE	
Transmitter Power (10 kW)	40 dbW
Transmitter Circuit loss	3 db
Antenna Gain (30 ft dish)	43.4 db
Polarization loss	3 db
Path loss (2500 n.mi. at 2101 MHz)	172.1 db
Receiver Antenna Gain	-3.0 db
Receiver Noise Figure	10.0 db
KT (T = 270°K)	204 dbW/Hz
Composite S/N Ratio	96.3 db/Hz
UP VOICE	
Composite S/N Ratio	96.3 db/Hz
Modulation loss	7.3 db
Bandwidth (20 kHz)	43 db
Received S/N Ratio	46 db
Desired S/N Ratio	10 db
Signal Margin	+36 db
UP DATA	
Composite S/N Ratio	96.3 db/Hz
Modulation loss	7.3 db
Bandwidth (20 kHz)	43 db
Received S/N Ratio	46.0 db
Desired S/N Ratio	13 db
Signal Margin	33.0 db

TABLE 6-1. (Concluded)

UP RANGE CODE	
Composite S/N Ratio	96.3 db/Hz
Modulation loss	12.1 db
Bandwidth (1.5 MHz)	61.8 db
Received S/N Ratio	22.4 db
UP CARRIER	
Composite S/N Ratio	96.3 db/Hz
Modulation loss	5.5 db
Bandwidth (800 Hz)	29 db
Received S/N Ratio	61.8 db
Desired S/N Ratio	12.0 db
Signal Margin	49.8 db

TABLE 6-2. MARGIN CALCULATIONS FOR THE S-BAND LINK

SHUTTLE TO MSFN	
Transmitter Power (20W)	13 dbW
Transmitter Circuit loss	6 db
Antenna gain	-3 db
Polarization Loss	3 db
Path Loss (2500 n.mi. at 2282.5 MHz)	172.8 db
Receiver Antenna Gain (30 ft)	44 db
Receiver Circuit loss	1 db
KT (T = 290°K)	204 dbW/Hz
Composite S/N Ratio	75.2 db/Hz

TABLE 6-2. (Concluded)

DOWN VOICE	
Composite S/N Ratio	75.2 db/Hz
Modulation loss	9.8 db
Bandwidth (20 kHz)	43 db
Received S/N Ratio	22.4 db
Desired S/N Ratio	10 db
Signal Margin	14.4 db
DOWN TELEMETRY	
Composite S/N Ratio	75.2 db/Hz
Modulation loss	9.8 db
Bandwidth (150 kHz)	51.8 db
Received S/N Ratio	13.6 db
Desired S/N Ratio	13 db
Signal Margin	0.6 db
DOWN-RANGE CODE	
Composite S/N Ratio	75.2 db/Hz
Modulation loss	18.7 db
Bandwidth (1.5 MHz)	17 db
Received S/N Ratio	39.5 db
DOWN CARRIER	
Composite S/N Ratio	75.2 db/Hz
Modulation loss	12.6 db
Bandwidth (800 Hz)	29.0 db
Received S/N Ratio	33.6 db
Desired S/N Ratio	12.0 db
Signal Margin	21.6 db

6.7 Automatic Checkout

Complete automatic checkout of the communication subsystem could involve signal simulators that surpass the complexity of the equipment being tested. As an example, consider the equipment required to check out an RF receiver. Assume that the level of testing or checkout is comparable to the unit acceptance tests. The minimum equipment required would be: an RF signal generator capable of being modulated and having a calibrated precision power level control, a modulation generator or oscillator, a distortion analyzer, a spectrum analyzer, noise figure meter, rms voltmeter, and a means of measuring error rates. This list of equipment seems excessive to be carried on the shuttle. A compromise would be to include the minimum equipment to reasonably assure that the component is operating satisfactorily. Another example is the checkout of the antenna system. Checkout of an antenna system internal to the shuttle requires a signal external to the shuttle. One either generates a signal external to the shuttle or uses the actual signal to be received as a test signal. The latter is recommended as a gross test. At this time it appears that only a gross onboard checkout of communication equipment is practical and that a detailed checkout should be performed on the ground where the adequate test equipment is available.

6.8 Data Bus Interface

The data bus will be utilized to change operating modes of the communication subsystem and for monitoring and checkout. In the past, operating modes were changed manually by a crew member. Examples of this are on/off function of receivers and frequency selection. Remote control of communication elements via a data bus represents a departure in element design and fabrication. Most items that will meet the electronic requirements now, will in all probability, have to be repackaged to interface with the data bus. A preliminary estimate of data to be received or transmitted via the data bus is shown on Figure 6-1.

7.0 Control and Display Subsystem

The control and display (C& D) subsystem for the shuttle must have the capability to support onboard checkout and autonomous operation without ground support. In addition it must provide controls and displays required for the vehicle to operate as a manned spacecraft and as an aircraft. The combination of these requirements into one system impacts the C & D requirement considerably.

Previous display panels in both spacecraft and high performance aircraft have occupied large area and volume and have been somewhat cluttered with electromechanical devices. The goal in designing the crew's console for the shuttle is to overcome these problems and provide the crew with a flexible control and display subsystem capable of combining the three major tasks of the display system and integrating man into the system with a minimum work load.

7.1 Control and Display Requirements

Basic control and display requirements or goals for the shuttle are as follows:

(1) Operation by a two-man crew; operable by one from either seat.

(2) Minimization of number of displays and controls.

(3) Elimination of circuit breakers.

(4) All-electronic display.

(5) Optimization of display area; clarity and comprehension of displayed data.

(6) Interface with a data bus and maximum utilization of digital information.

(7) Complete display flexibility and adaptation to vehicular system changes.

(8) Integration of the three display tasks, outlined earlier, into a single system.

To finalize the requirements will require an indepth evaluation of the respective systems for data presentation requirements and considerable interface with human factors personnel and prospective crew members. However, the goals of the control and display subsystem are sufficiently defined to evaluate and define the equipment approach.

No attempt will be made in this study to lay out the control and display panel; any figures or comments relative to equipment location and form of material presentation are strictly for the purpose of illustrating equipment capabilities and fulfillment of the display task. Actual location of individual components, selection of display symbology and form of presentation is another study requiring extensive evaluation by human factor engineers and flight personnel.

A simulation facility for mock-up of selected hardware for crew and human factor evaluation is deemed highly desirable. In such a facility, hardware location and format of data can be varied to produce the most desirable and optimum combination to meet mission requirements and crew acceptance.

At this time the degree of automation of the shuttle has not been defined. As for the controls and displays, the same requirements will exist for a manned mission regardless of the extent of automation. In a completely automated system the C & D requirements will be necessary to provide crew management of the vehicular system and control in case of an emergency.

7.2 Display Task

A preliminary list of parameters requiring display, compiled by LMSC, totals approximately 500 display modes and 300 control modes [1, Vol V]. This is only a preliminary functional list where many of the items have multiple outputs, and omissions or redundancies may be present. It is used only as a guide to establish a size for the C & D task. Utilizing these figures, a capability to handle approximately 1000 C & D functions is considered necessary. Working under the design ground rules previously outlined, one can easily see that neither console volume and area nor the envelope defined by human viewing and reach limitations can accommodate individual dedicated instruments to fulfill this task. Integrated electronic displays, programmable and controllable, are therefore desirable to overcome this problem. In addition, simplification of information presentation and minimization of crew workload can also be realized.

7. 2. 1 Displays

Display systems take many forms and use a wide variety of techniques to provide the most effective transmission of information to the operator. Regardless of the method, display systems may be classified into five basic categories: continuous (analog), digital, symbolic, pictorial, and directed motion. Although the final design may include a mixture of these types, the pictorial display presents a greater degree of flexibility and display capability than the others.

Electronic displays fall into three general categories: X-Y flat panels, projection, and direct view cathode ray tube (CRT) displays. Of these, the direct view CRT is the most promising device for the era of the space shuttle. Devices such as plasma, liquid crystal, and light emitting diode matrices in the X-Y flat panel category have characteristics that are desirable but their state of development restricts them from consideration for use in the early 70's [2, Vol III, and 6]. Photochromic film, light valves with Schlieren optics, and laser projector devices were also reviewed for possible use; however, they too were deemed unacceptable due to their early state of development. Table 7-1 [2, Vol III] gives some of the characteristics of various display devices and their development state. The direct view CRT is the most versatile of the display devices presently available. The vast amount of technology and experience through the years on CRTs makes it an attractive selection for the display media for the shuttle. In addition, the CRT is rugged and reliable while its associated electronics can be built compactly with relatively low power requirements.

Display system capabilities can be categorized into housekeeping and operational functions. To display the information required of the various systems, the C&D equipment must be capable of displaying radar, TV, alphanumeric, and analog data. Review of these data requirements, the flight mode in which they are required, and the form of presentation must be evaluated simultaneously to determine the number and type of displays required.

Available technology in display devices includes the CRT computer/man interface, vertical situation displays (VSD), horizontal situation displays (HSD), heads up displays (HUD), and electroluminescent scale scribers, presently being used by high performance military aircraft and being evaluated for use on commercial aircraft. New technologies include such devices as the electronic attitude director indicator (EADI) and multimode displays presently being evaluated for future military and commercial aircraft such as the SST, F-15, and DC 10. Although these devices are designed specifically

TABLE 7-1. CHARACTERISTICS OF VARIOUS DISPLAYS

Display Type	Mode	Brightness (foot-Lamberts)	Resolution (line/in.)	Weight (lb)	Average Power (Watts)	Reliability	Probability of Availability in 1972 (%)
Plasma Panel	TV	300	50	30	100		60
	Random	500	50	25	55	Unk.	75
	Alpha-Num	500	50	25	55		75
Light Emitting Diodes	TV	200	40	50	1500		
	Random	200	40	40	750	High	40
	Alpha-Num	200	40	35	500		
CRT	TV	200	100	15	25		
	Random	50	100	70	320	High	100
	Alpha-Num	100	100	70	320		
Storage CRT	TV	2000	40	75	400		
	Random	2000	40	75	350	Medium	100
	Alpha-Num	2000	40	75	350		

for use in aircraft and lend themselves well to certain phases of the shuttle mission, they can also be adapted for use during the spacecraft phases of the shuttle mission. Table 7-2 shows the adaptability of each of these instruments to the different modes of the shuttle mission.

The raster scan CRT display with calligraphic capability is the most versatile readout available at this time and is not likely to be replaced in the near future. Its capabilities meet the needs of a variety of diverse applications and can be adapted for use in EADI, HUD, HSD, and multifunction display (MFD) systems. With the addition of a radar scan converter, the resultant multimode display is capable of presenting radar data, television, infrared, laser scan, and synthetically generated symbols on a common readout.

Several methods exist to generate the data for display on the CRT of the above devices. These methods will be covered in Appendix C. Of the methods, the optronic (rear projection CRT) is selected for the HSD to eliminate computer map generation requirements and increase map resolution. Some advantages can also be realized in the MFD utilizing this technique.

The EADI does not require the rear projection capability in its normal mode of operation. The optronic type tube is recommended for the EADI to achieve more display flexibility and meet redundancy requirements without additional displays.

7.2.2 Display Requirements

The following is a preliminary list of display and control functions required in the crew compartment:

Guidance and Navigation Requirements

All Phases of Flight

Mission time
GMT and EST
Latitude
Longitude
Attitude; attitude rate
Velocity
Path angle
Azimuth

TABLE 7-2. DISPLAY USAGE

MISSION PHASE / DISPLAY TYPE	PRE LAUNCH	LAUNCH & ASCENT	ORBIT INSERTION	RENDEZVOUS	DOCKING	ORBIT STAY	DEORBIT	RE-ENTRY	APPROACH	LANDING
EADI		• MONITOR MANEUVERS	• PREP FOR RETRO- • ORBIT ATTITUDE MONITOR	• APPROACH • RENDEZVOUS	• DOCKING MANEUVERS		• MONITOR MANEUVERS	• MONITOR GNC • MONITOR MANEUVERS	• MONITOR MANEUVERS • ILS • PILOTAGE	• MONITOR MANEUVERS • ILS • PILOTAGE
HSD	• CHECK LIST	• COMMUNICATION CUEING	• MERCATOR MAP • GROUND TRACK • ORBIT TRANSFER SETUP	• COMMUNICATION CUEING	• RADAR	• GROUND TRACK • ANTENNA POINTING	• ORBIT TRANSFER SETUP • GROUND TRACK	• ENTRY ENERGY MANAGEMENT • TRAJECTORY MANAGEMENT • GROUND TRACK • FOOT PRINT MAP	• APPROACH & LANDING AREA MAP • STEERING COURSE & RANGE • COMMUNICATION	• AREA MAP • COMMUNICATION CUEING • STEERING COURSE & RANGE • RADAR CUEING
HUD		• MONITOR MANEUVERS		• RENDEZVOUS • APPROACH	• DOCKING • UNDOCKING			• MONITOR MANEUVERS	• MANEUVER LIMITS • ILS • PILOTAGE	• MANEUVER LIMITS • ILS • PILOTAGE
MULTI FUNCTION	• CHECKOUT • CONFIGURATION • LAUNCH DATA • CHECKLIST	• MONITOR LAUNCH CORRIDOR • CHECKLIST • RADAR	• TV DISPLAY • CHECK LIST • MALFUNCTION ISOLATION • TARGETING	• TV DISPLAY	• TV MONITOR	• TV DISPLAY • CHECKOUT • CHECK LIST • MALFUNCTION ISOLATION	• TV DISPLAY • CHECKOUT • FAULT ISOLATION	• TV DISPLAY • FOOT PRINT MONITOR • REFERENCE MONITOR • PROPULSION PERFORMANCE	• DECENT CORRIDOR MONITOR • TV DISPLAY • RADAR	• RADAR • PROPULSION PERFORMANCE
VSD		• MONITOR MANEUVERS • PILOTAGE		• MONITOR MANEUVERS	• DOCKING MANEUVERS • UNDOCKING MANEUVERS			• MONITOR MANEUVERS	• MANEUVER LIMITS • ILS • PILOTAGE	• HEADS DOWN LANDING • ILS • MANEUVER LIMITS
ELECTRO-LUMINESCENT									• ENGINE PERFORMANCE • CAUTION & WARNING	• ENGINE PERFORMANCE • CAUTION & WARNING

Angular reference (3 axes)
Configuration status
Attitude error (3 axes)
Flight mode indication
Hydraulic pressure
RCS ΔV remaining
Angle of attack
RCS operation.

Boost Phase

Countdown events (engine cutoffs, starting sequence, ignition, etc.)
Velocity to be gained by cutoff
Main engine actuator positions.

Orbit Phase

Change in translational position
Change in translational velocity
Vehicle ephemeris
Target ephemeris
Closing/separation requirements (plane changes, angles, ΔV's, phasing, and times)
Event countdown (engine ignition, cutoffs, etc.)
ΔV countdowns
Navigation update sensor status
Indicated inertial navigation error and required update frequency.

Docking Phase

Range; range rate
Clock angle; clock angle rate
Cone angle; cone angle rate
Closing/separation requirements (line of sight pitch and yaw, ΔV, timer, etc.)
Event status (time till contact, control event, latching event, etc.).

Reentry and Approach

Energy management data
Ground track
Magnetic North
True North

Vortac acquisition
Range to outer marker
Jet engine data (time to ignition, event, fuel status, etc.)
Air speed
Deceleration
Skin temperature
Altitude above ground/runway
ILS acquisition
Aerodynamic surface actuator position.

Landing

Runway profile
Distance to runway
Glide slope deviation
Localizer deviation
Ground winds
Air speed
Crab angle
Marker beacon passage indication
Air temperature
Brake temperature
Ground speed.

Communication

Power control
Receive/transmit
Frequency selection
Volume

} For each piece of communication equipment

Electrical Power

Ac power (voltage, current, and frequency monitors)
Dc power (voltage and current indicators)
Ground supply indicators
Battery charge current
KW-hr counter
Fuel Cell
- Efficiency
- Temperature
- Chemical status

Heater status
KW-hr capability remaining
Energy remaining
Power system control
Status
Inverter status
Alternator status
Overload condition; ac and dc
Power switching status and control
Auxiliary power system monitoring and control
Power distribution (circuit breaker status).

Propulsion

Propellant storage
Fuel pressure
Fuel quantity
Oxidizer pressure
Oxidizer quantity
Flow rates
Combustion chamber
Pressure
Temperature
Stability
Turbine inlet temperature
Fuel inlet temperature
Oxidizer inlet temperature
Thrust level
Percent power.

Vehicle Configuration

Position of equipment such as jet engine, speed brakes, elevons, cargo doors, landing gear deployment and doors, antenna deployment, hatches, airlocks, etc.

Environmental Control

Cabin pressure
Oxygen partial pressure
Cabin temperature
Flow rates

Tank pressures and quantities
Coolant system monitors
Miscellaneous controls and check points.

7.3 Control and Display Concept

The proposed functional control and display system chosen for this conceptual design is depicted in Figure 7-1. Figure 7-2 shows a block diagram of a programmable display system. It consists of two EADI's, two HSD's, a multifunction display, two groups of vertical scale indicators, and a group of dedicated instruments. Two HUD's are shown but, as will be explained in Paragraph 7.4.2, their presence in the final design is questionable. This selection was based on the display requirements known at this time and instrument capabilities outlined in Paragraph 7.2. The two crew stations are identical to satisfy the requirements for operation by one man plus making it possible to display to the second crewman the same information as the pilot sees during periods of critical activity. During other mission phases, the second crew station displays can be utilized to perform subsystem checkout, malfunction isolation, configuration monitoring, and other housekeeping activities.

The entire repertoire of displays can be programmed to display different information during various modes of shuttle flight. Critical events or status can be programmed to take precedence over a noncritical display. Other modes may be controlled manually by the crew, or the formats can be changed automatically by the computer. The programs controlling the format and the format generation can be changed as display and equipment needs dictate without disturbing the display hardware.

Programmable redundancy can be incorporated to improve reliability and conform to the reliability requirements of the shuttle. Voting can be utilized in the display data processor to determine major display element failure and provide automatic switching of redundant components or inform the crew of required action.

Checkout and fault isolation of the C&D system falls into two categories. In an automatic mode, the points listed in Section 7 of Appendix A can be monitored for confidence checks and preflight checkouts. For fault isolation and system performance, pictorial display systems exhibit a unique capability. By examination of a self-generated test pattern, an operator can detect any malfunctioning or degrading portion of the system. This technique is valid only in cases where sweep circuits and the CRT tube itself are not involved. In these instances the pattern can be switched to another display, thus isolating the malfunction.

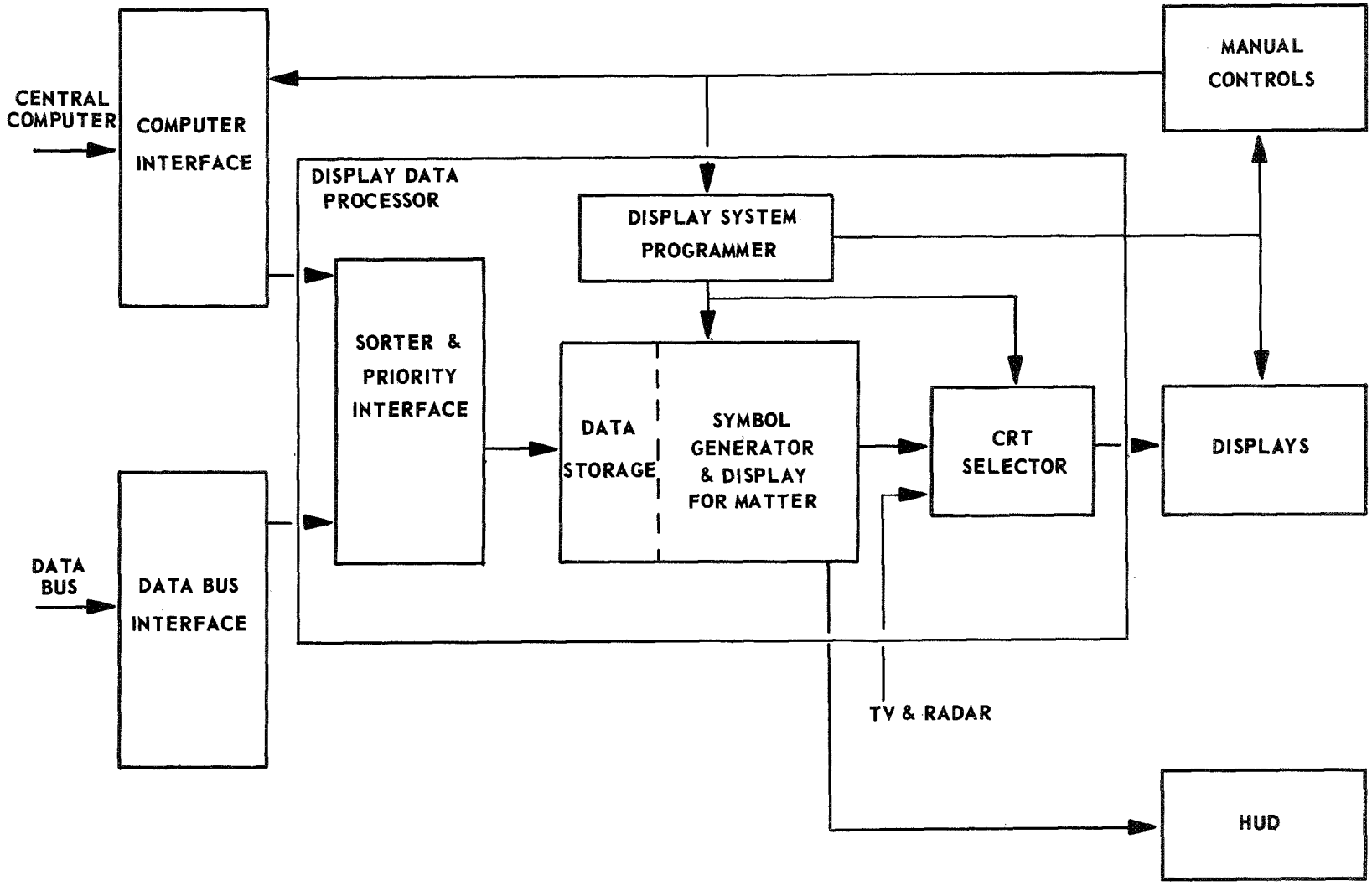


Figure 7-1. Control and display functional block diagram.

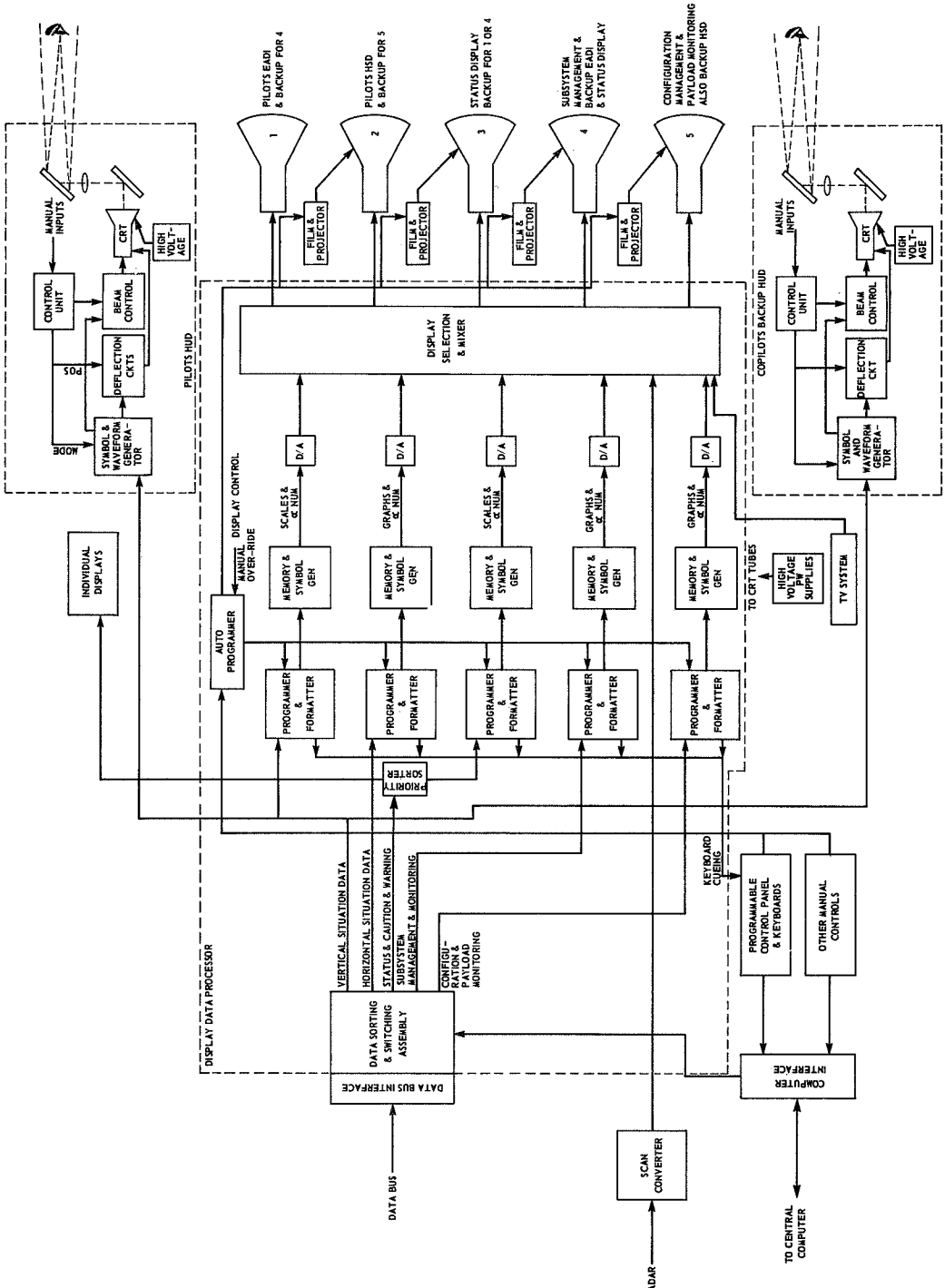


Figure 7-2. Programmable display system.

The block diagram of the programmable display system shown in Figure 7-2 is of the conventional concept. This diagram was presented to show the flow of data from data bus to display; however, in implementation it would be desirable to use a computer-managed display. A computer-managed system eliminates the requirement for separate electronics, memory, formatter, and controls for each display system by combining all these functions in a computer-managed unit. The advantages of such a system can be realized in the following:

(1) Reduction in —

- Weight
- Power
- Space
- Cost
- Complexity

(2) Improved —

- Reliability
- Flexibility
- System Management

(3) Capability to handle large amounts of data and sensor information which must be displayed.

The block diagram of such a system is shown in Figure 7-3.

Table 7-3 gives typical power and weight requirements for the display system shown in Figure 7-2. These figures are based on dedicated electronics for each display and do not include redundancy other than the back-up capability of dual displays. The computer-managed display system shown in Figure 7-3 would reduce the power and weight requirement to approximately 2 kilowatts and 350 pounds with a sizable decrease in volume. A substantial overall reduction is not possible since indicator weight and power are a high percentage of each system requirement. Multiple redundancy and increased

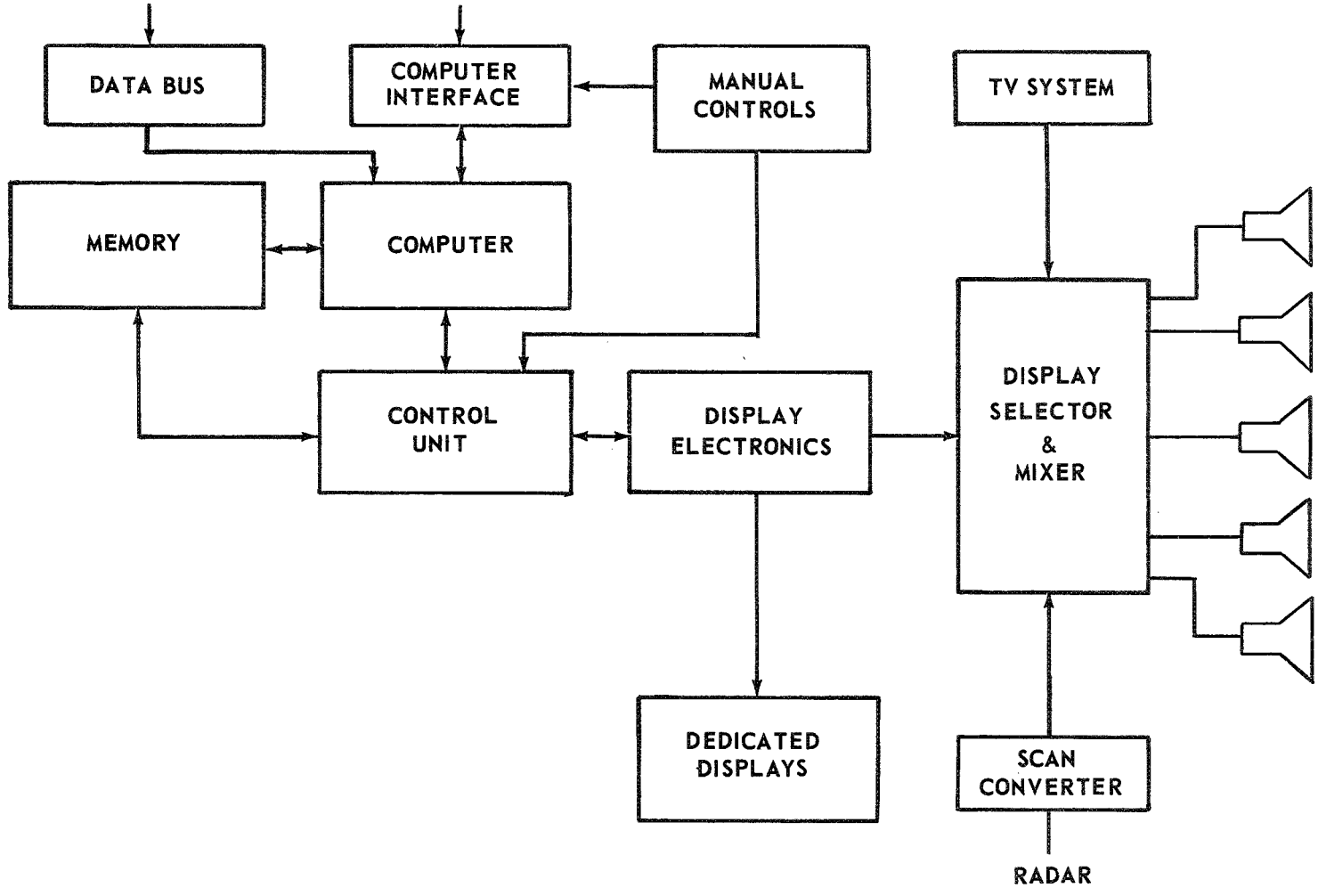


Figure 7-3. Computer-managed display system.

flexibility are accomplished through utilization of the computer-managed system. These estimates include neither control input devices (yoke, pedal, hand controller) nor communication equipment located in the control and display area.

The concept presented here is that proposed for the orbiter vehicle. Although the dedication of some of the devices would be different for the booster stage of the shuttle, no significant difference in the C&D equipment is envisioned.

TABLE 7-3. WEIGHT AND POWER ESTIMATES FOR DEDICATED DISPLAYS SYSTEM

Item	Quantity	Power (Watts)	Weight (lb)
EADI			
Indicator	2	300	32
Symbol Gen.	2	400	70
MFD			
Indicator	1	320	20
Symbol Gen. and Processor	1	135	32
HSD			
Indicator	2	640	92
Processor	2	120	28
Symbol Generator	2	150	36
HUD (Viewing Screen)	2	140	40
Scan Converter	1	150	30
Display Selector and Mixer	1	35	10
Interface Equipment	1	40	30
Indicators		60	10
Total		2490	430

7.4 Selection and Utilization of Proposed Equipment

The rationale for selection and utilization of each type of equipment proposed in this conceptual design is covered in the following paragraphs.

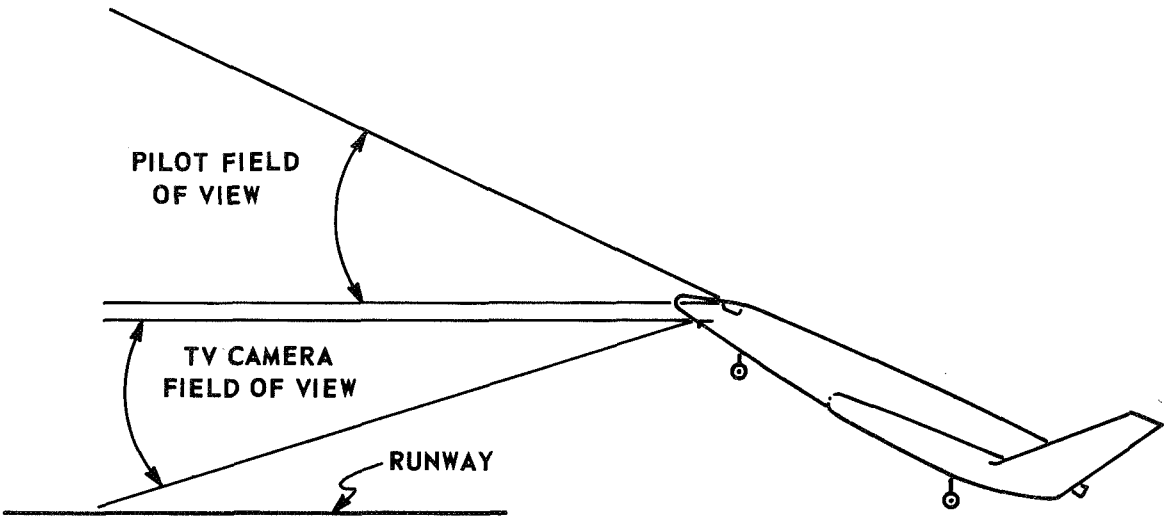
7.4.1 Electronic Attitude Director Indicator (EADI)

The EADI is a prime example of the capability to reduce clutter and number of dedicated instruments in the control console by use of electronic display devices. The EADI integrated into one instrument the electromechanical 8-ball, altitude, vertical speed, airspeed, and Mach number instruments. Through the use of electronically generated symbology, it provides a simultaneous display of critical flight information; heading, airspeed, attitude, altitude, and command and rate information in clear, immediately recognizable form. In addition an EADI can display TV and radar inputs. The on-sight availability of these data eliminates the need for mental calculation, interpretation, and integration by the operator. Complementary information is integrated, reducing the array of separate instruments which continually vie for the crew's attention. Table 7-2 lists the proposed use of the EADI for various phases of the shuttle mission. In summary the EADI offers the following advantages: (1) single-instrument replacement of several dedicated devices; (2) multimode operation to include EADI symbology superimposed on an external TV picture; (3) scale factor and symbology variation to optimize for each mission phase; and (4) acts as a backup for other CRT display devices.

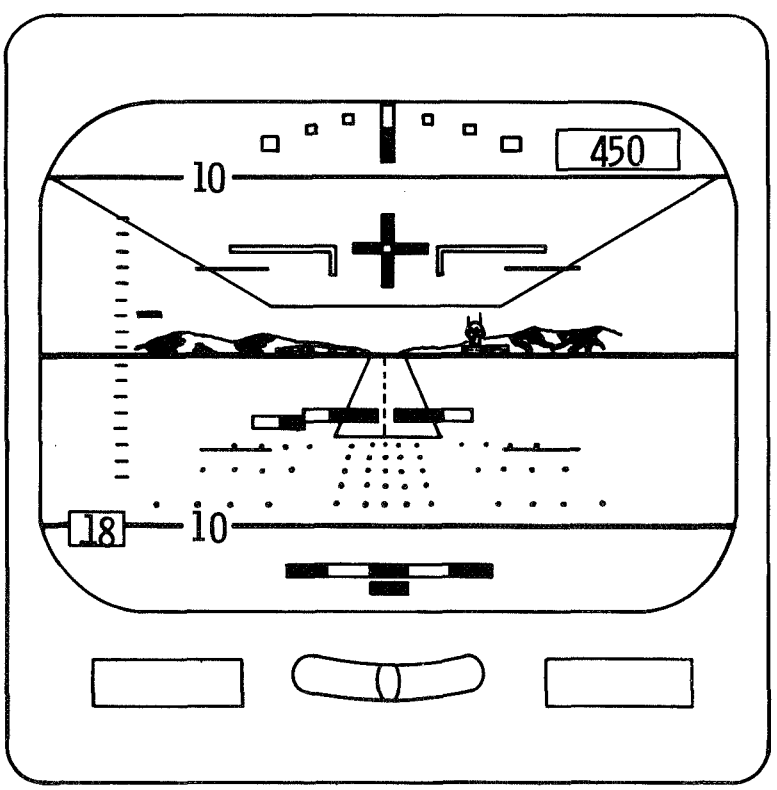
In reference to the multimode operation (2 above), such a system is being flight tested by Boeing in a 707. This application is discussed in the HUD rationale and is a contingency consideration for use during landing. This decision will be part of a trade-off study and will primarily be dependent on vehicle configuration. Figure 7-4 depicts this mode of operation.

7.4.2 Heads Up Display (HUD)

The primary phases for utilization of a HUD in the space shuttle are rendezvous, docking, and landing. The HUD does not provide any information that is not available on a vertical situation display (VSD) or an EADI; but it offers the attractive advantage of displaying data superimposed on the view of the external environment. Appendix C includes a breakdown and operational description of the HUD. The display of data on the visual reference permits the crew member to see the information pertinent to the maneuver being performed without removing his eyes from the actual reference. Under conditions where the vehicle is moving at a high rate of speed this is a distinct advantage since considerable distance can be traveled during glances between the instruments and the real image. In addition, the operator is subject to disorientation during these periods. The HUD overcomes these disadvantages.



a. ILLUSTRATION OF OBSTRUCTED VISION



b. EADI/TV DISPLAY

Figure 7-4. Utilization of EADI/TV mode.

7. 4. 2. 1 Rendezvous and Docking Phase

During the rendezvous and docking maneuver the target approach data would be superimposed on the visual reference of the vehicle with which docking is being attempted. Information available to the crewman on the display would consist of target range/range rate, target azimuth/elevation, cross-hair reference (docking roll), and ΔV status.

7. 4. 2. 2 Landing Phase

Limited visibility due to weather is alleviated by the HUD by projection of analog runway bars and landing approach data on a viewing screen in front of the pilot. The result is that the data and symbols appear to be superimposed on their real image counterparts as viewed through the windshield. A smooth transition is made from limited visibility to acquisition of runway by visual sighting without the necessity for scanning multiple instruments and the windshield for visual contact with the ground.

Data presented on the HUD in the landing phase include the following:

(1) Analog runway bars

(2) Pitch and bank information

(3) Altitude and Altitude rate

(4) Roll angle

(5) Airspeed.

The actual incorporation of a HUD in the space shuttle is questionable at this time. Its use during landing may be questionable, depending on the configuration of the vehicle and its attitude with relation to the runway during the landing phase. Some configurations presently under consideration have attitudes during landing that indicate the ground visibility will be minimal. If visibility is limited to the point that the crew cannot see the runway through the windshield, the data presented on the HUD will be of little use since it can be obtained on the EADI. If this is the case it is doubtful if HUD usage during docking will justify its incorporation into the C&D system.

The commercial division of the Boeing Company has been successfully landing a 707 using a multimode display as the pilot's only source of information concerning vehicular attitude during landing. The display consists of a TV picture of the view normally available from the cockpit superimposed on an EADI display. Several test flights have been conducted with favorable comments from test pilots. The display offers basically the same advantage as the HUD in that the pilot can concentrate on the display and not be required to glance away at any time during the entire landing operation. Application of a similar system on the shuttle seems highly probable, especially if the landing configurations under present consideration are implemented.

7.4.3 Horizontal Situation Display (HSD)

The horizontal situation display offers the capability to display useful information concerning the vehicle situation in the horizontal plane in all phases of shuttle flight from launch to touchdown. The HSD also offers the capability to display check lists, procedural instructions, maintenance information, airport traffic procedural diagrams, etc., during other periods of shuttle activity. During periods of minimal utilization or during emergencies, the HSD can be used either as a backup for other CRT displays or to display information supplementary to that on other displays, such as during checkout or fault isolation activities.

The HSD is primarily a moving map display and can be implemented in several ways as discussed in Appendix C; however, for the purpose of this conceptual design the rear projection CRT technique will be utilized. It consists basically of a multipurpose CRT display with rear projection capabilities which presents data stored on film superimposed on CRT electronically generated information. With these capabilities, alphanumerics and navigational symbols can be positioned on the projected data to show vehicle position relative to displayed data to complement it with information such as radio frequencies, runway heading, terrain elevation, etc.

Typical usage of the HSD in the shuttle operation is as follows:

(1) Prelaunch and launch

- Check list
- Countdown status
- Ground track during launch

(2) Orbit track

- Mercator map with ground track overlay.
- Guidance and navigation aids for orbit transfer setup.
- Communication cueing and antenna pointing information.

(3) Reentry and approach

- Energy management corridor with position and directed motion
- Trajectory management
- Range/crossrange footprint map (Fig. 7-5)
- Approach and landing area maps
- Steering course and range information

(4) Landing

- Runway pattern
- Vehicle situation relative to runway.

7.4.4 Multifunction Displays (MFD)

The size and complexity of the space shuttle will place additional demands on the crew in terms of their anticipatory and decision-making capabilities. The present state-of-the-art display and computer system can provide the crew with processed information which will enhance these capabilities while simultaneously decreasing workload. The net effect will be improved overall operational efficiency and flexibility of the vehicle. The MFD envisioned to aid in this task is a general-purpose, time-shared, optronic (rear projection CRT) display driven by data from an onboard computer, presenting information needed to meet these objectives.

Some of the applications of a MFD for the shuttle are as follows:

(1) Presentation of optimum flight profile information and energy management parameters.

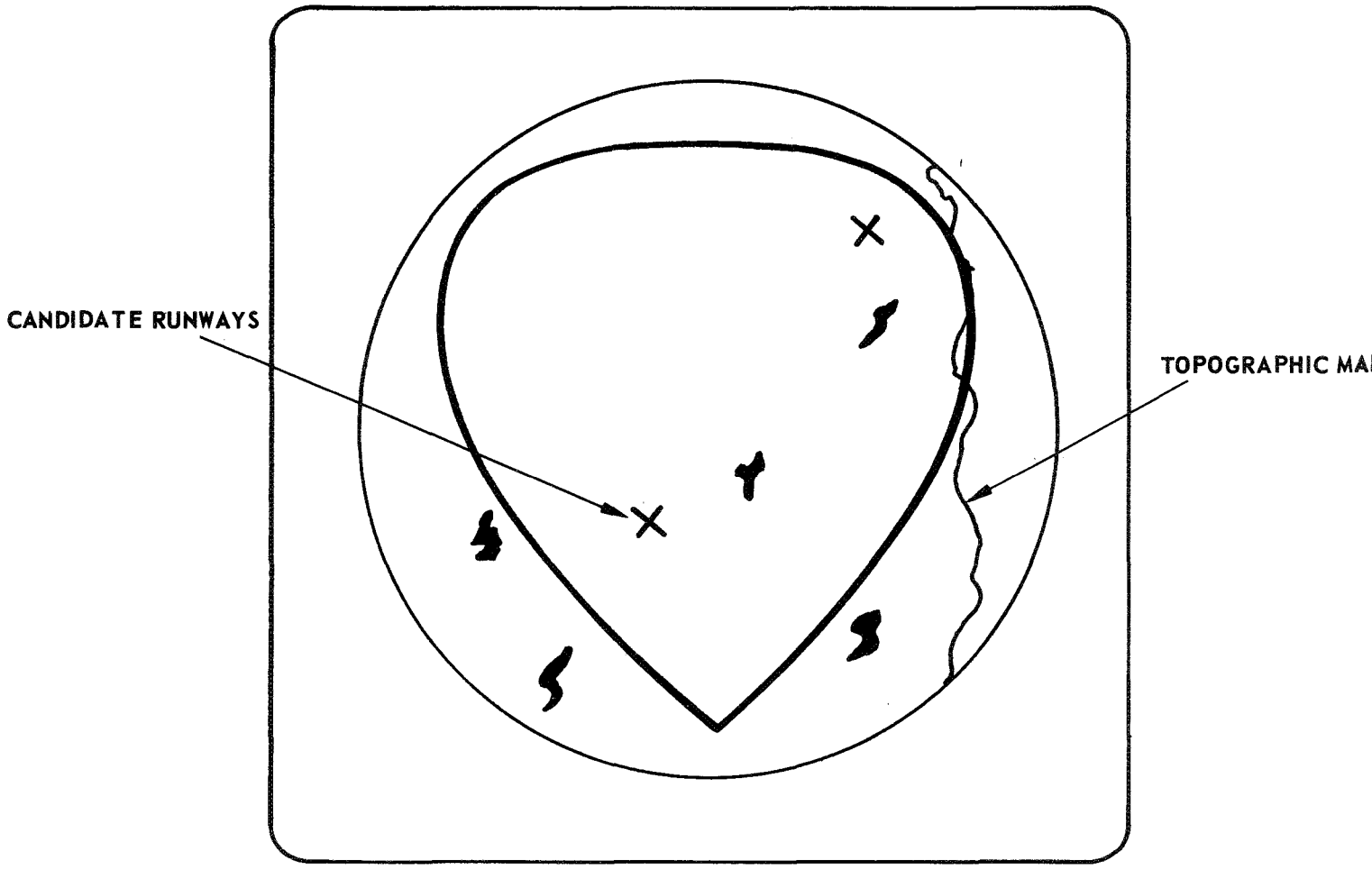


Figure 7-5. Instantaneous range and crossrange capability.

(2) Presentation of vertical profiles for ascent and descent corridors.

(3) Presentation of situation data from onboard systems for crew monitoring of operational performance.

(4) Presentation of check lists, emergency procedures, caution and warning data.

(5) Aid in crew/computer interface when entering new data into the computer and determining alternate course of action.

(6) Increase reliability of display system by acting as backup display for EADI, HSD.

A typical MFD block diagram is shown in Figure 7-6. The electronics of such a system can be integrated into the shuttle display control system as shown in Figure 7-3. In this configuration it is envisioned to utilize graphics, alphanumerics and special symbols to present operational data such as the following:

(1) Vertical navigation aids

- Ascent profiles
- Emergency descent profile
- Range and cross range landing capabilities
- Descent profiles

(2) Fuel management

- Center of gravity
- Propellant utilization information

(3) General information

- Time references
- Digital display of any flight data

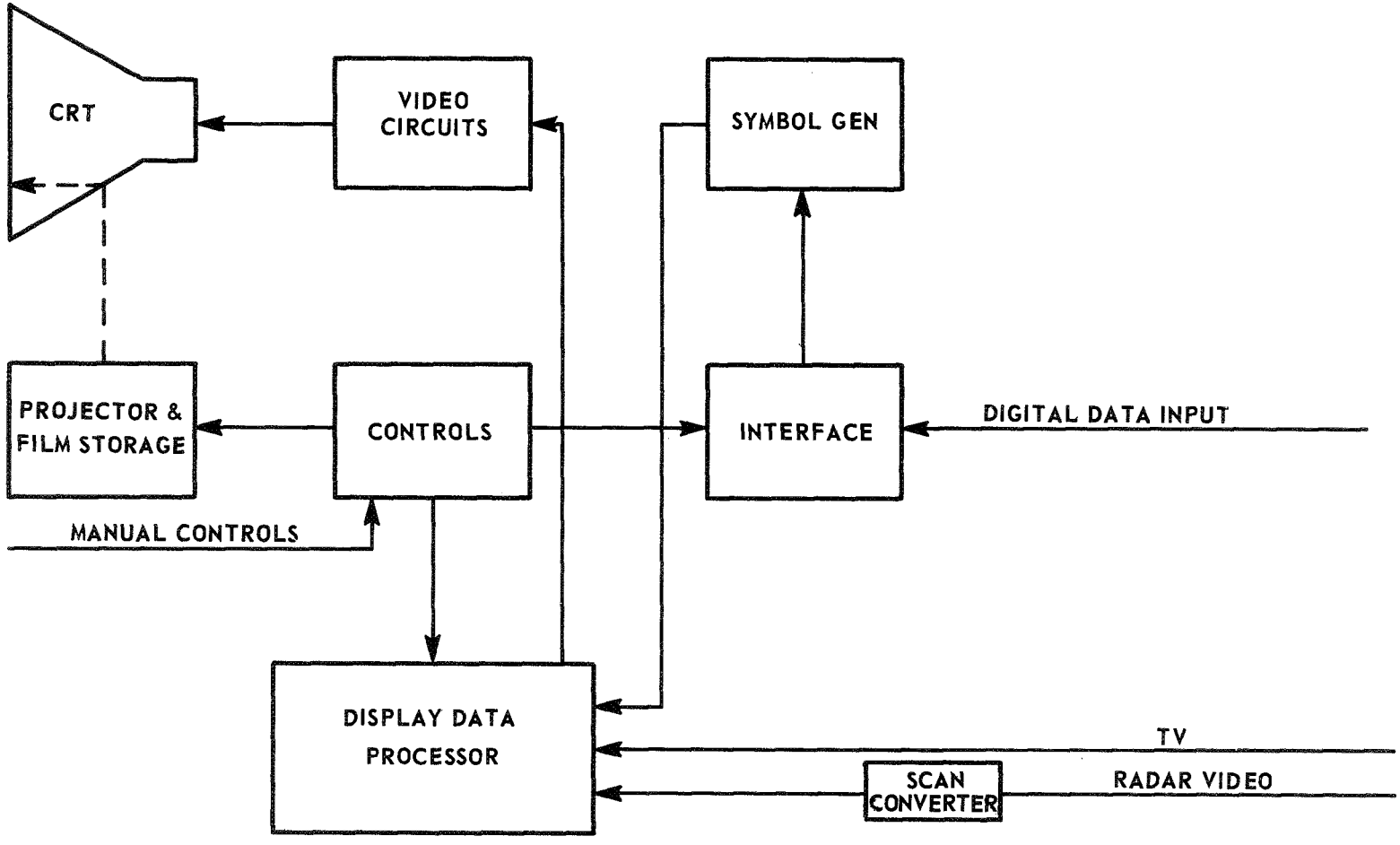


Figure 7-6. MFD functional block diagram.

- Check lists
- Closed circuit TV
- Radar data

(4) Safety aids

- EADI backup
- Engine restart envelope
- Emergency procedures
- Aerodynamic and heating loads information
- HSD backup
- Subsystem failure display.

Elimination of circuit breakers as part of the equipment located in the C&D area was listed as a goal of the shuttle design. Actual implementation of circuit breaker technology used in the shuttle is covered in the power section of this report. However, monitoring and some form of control must be provided by the C&D system. The monitoring task may best be performed by a programmable display operating in a MFD mode. Reset capability can be provided via the operating keyboard and the data bus. In this mode of operation, the status of all breakers can be monitored on demand by the operator. During other phases of the mission the operator would be alerted only when a circuit breaker trips. Once alerted, the operator can request a circuit breaker display upon which he is informed of which circuit breaker tripped, its function and power source, its location, and the address code for resetting (Fig. 7-7).

Actual selection of functions to be performed by the MFD and form of display will require further analytical and simulator studies to ascertain optimum performance.

Figures 7-8 through 7-10 illustrate the type of information that can be displayed on such a device. Neither the format nor the function being performed are final designs, but are presented only to illustrate the display capabilities. It should be further pointed out that the entire display need not be dedicated to a particular display function at any given time. Different portions of the screen can display nonrelated information simultaneously.

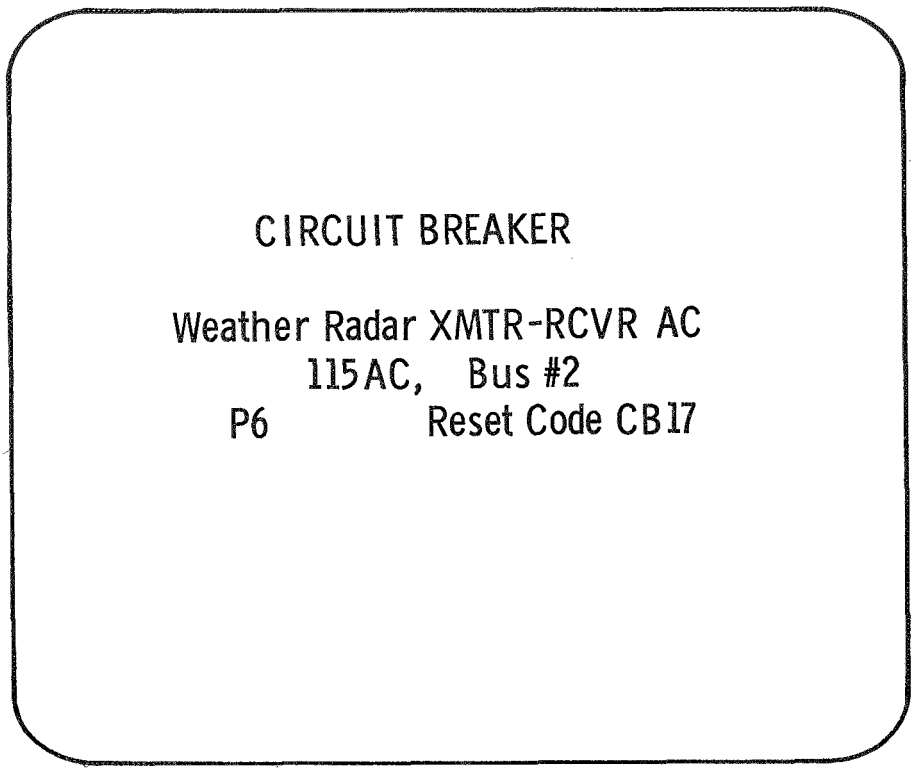


Figure 7-7. Typical circuit breaker format.

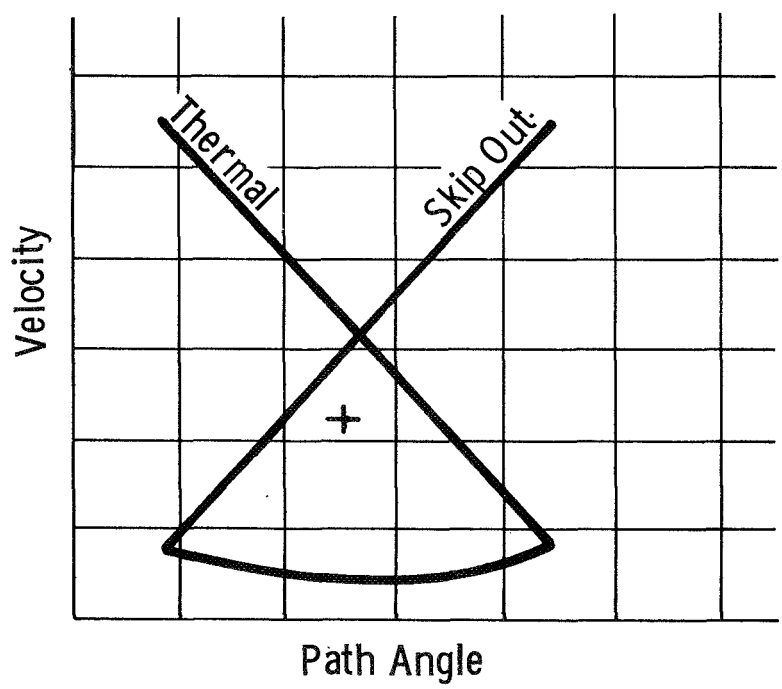


Figure 7-8. Reentry corridor.

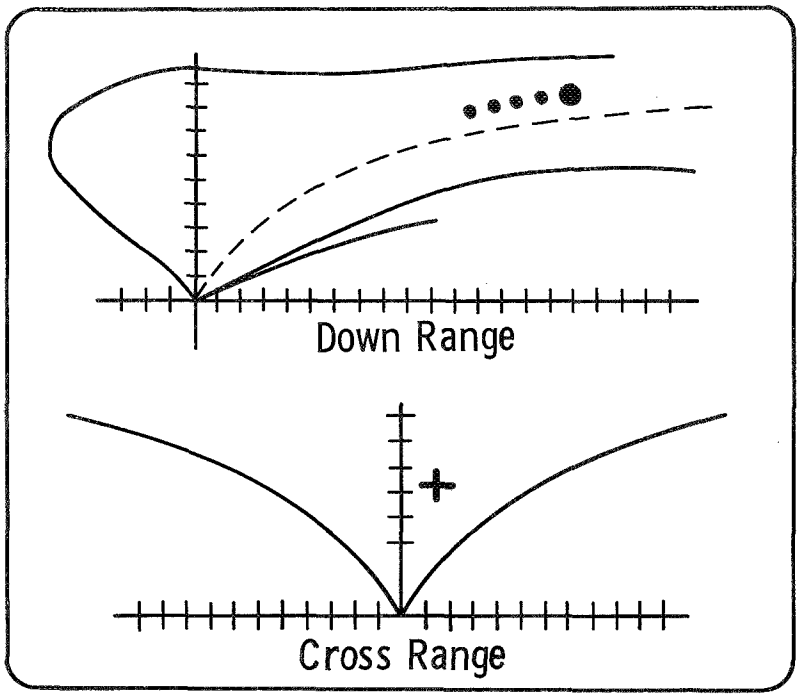


Figure 7-9. Cross-range and down-range landing corridors.

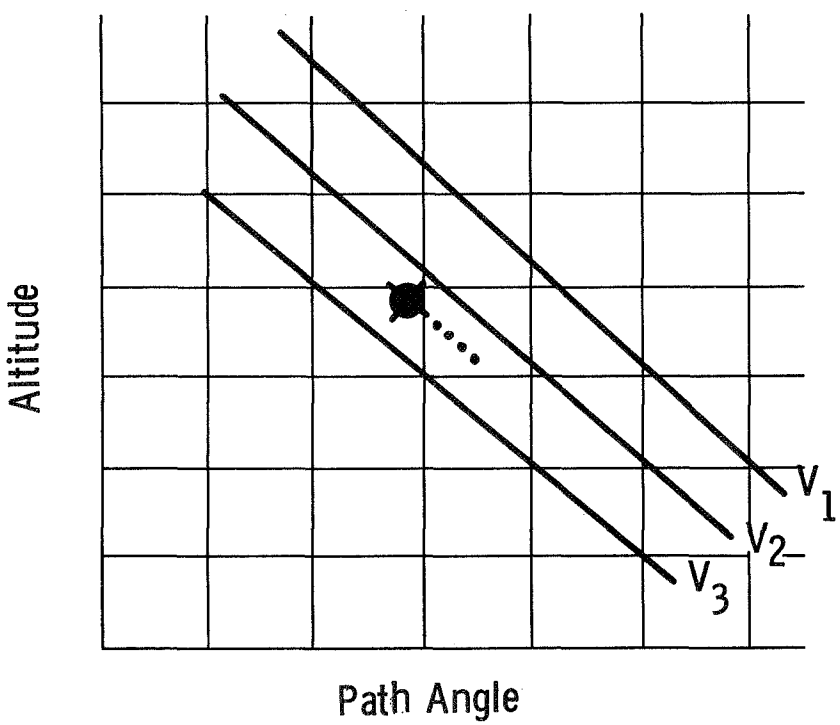


Figure 7-10. Energy corridor.

7.4.5 Electronic-Moving Bargraphs

Measurements requiring continuous or near continuous display may be better displayed on dedicated instruments. Vertical scale instruments have recently found favor for readouts such as temperature, pressure, flow rate, quantity, ratio, etc. Of these instruments, electroluminescent scale scribers and light emitting diode-moving bargraphs are the two prime candidates for consideration. These instruments reduce panel area, eliminate parallax, and can be driven directly from a digital interface.

By proper programming, scales and function being displayed can be changed. This technique is accomplished by changing legend and/or scales which are visible only when lighted. This capability could be utilized for such systems as the engines where different engine systems are used in the different phases of the mission.

7.4.6 Other Devices

The preceding material has dealt primarily with visual display concepts. Other types, such as auditory, may be found useful in the final design. One specific use of aural devices is caution and warning. Recorded messages are another consideration, especially during preflight checkout.

7.4.7 Reprogrammable Switches

A significant problem in display technology has been the interaction of man with computers. The interaction requirements are further expanded when the computer is used for multiple purposes, as is the case for the space shuttle. A solution to this problem is the use of reprogrammable switches.

Dedicating a separate switch to every function overcrowds panels, causes operator difficulties, and is wasteful. It is more effective to perform the multitude of switch functions with a small number of switches. To make this possible, each switch must be reprogrammable to perform several functions; and, more importantly, all the choices that the pilot might wish to make at any one time must be available to him.

Implicit in this concept is that, at any one moment, the pilot's actions are by nature limited to a relatively small number of options, even when he performs complicated tasks. Complicated tasks usually are separated into a sequence of simpler steps. Real help can be provided in the form of cues that inform the pilot of his options.

Several solutions to reducing the number of switches use this concept. In each, the switch function is changed by having the switch interact with the computer. Switch closures are coded inputs to the computer. When an operator depresses one of the switches, a coded pulse is transmitted to the computer. The computer identifies the switch position, transmits a control signal, then reprograms the switch functions. Three techniques of accomplishing this task follow:

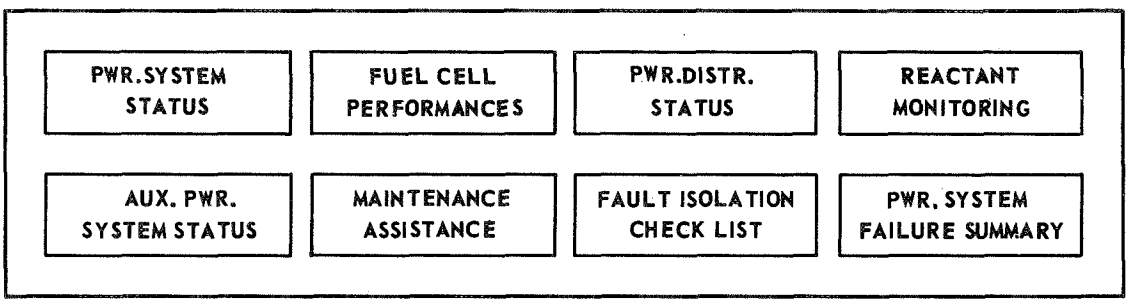
(1) Display on Face of Pushbutton — The pilot must know what functions the switch will perform. This becomes convenient for him if the label on, or directly next to, the switch changes with the changing switch function. Figure 7-11 (a) shows an example of pushbuttons with their programmable functions displayed directly on the button faces. This method uses a film slide that contains many individual messages with a bulb or optics behind each message. The switch legend is changed by activating the appropriate bulb, or moving the optics, or both. The message is projected onto a screen on the front portion of each switch. Reprogrammable function switches using the projection-filmed message are currently being used with 24 and 48 individual messages per switch.

(2) Display Next to Pushbutton — Figure 7-11 (b) shows a second and highly flexible display method. The display is immediately next to the pushbuttons. Each message is close enough to its corresponding pushbutton so that the face of the button can be left blank. The display medium can be cathode ray, plasma, or any other totally reprogrammable type. The quantity of messages is not limited as in the first method.

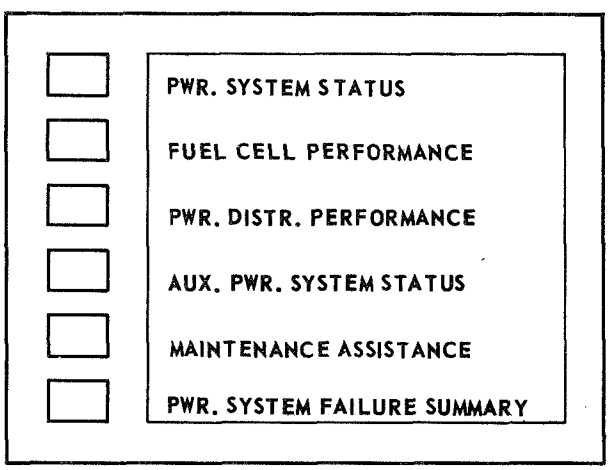
(3) Keyboard Concept — Functionally less convenient for rapid response actions is the keyboard technique shown in Figure 7-11 (c) which identifies the changed switch function on a display near the pushbutton. The pushbutton has a fixed legend on its face, usually an alphanumeric or symbolic code. The nearby programmable display, such as a CRT, or some other digital device, shows the switch legend with the switch function spelled out next to it.

In addition, the standard computer interface keyboard offers the maximum flexibility, but is least desirable as far as the man-machine interface is concerned. However, incorporation of such a keyboard is mandatory to provide the computer interface for instructions and the nonfrequent address.

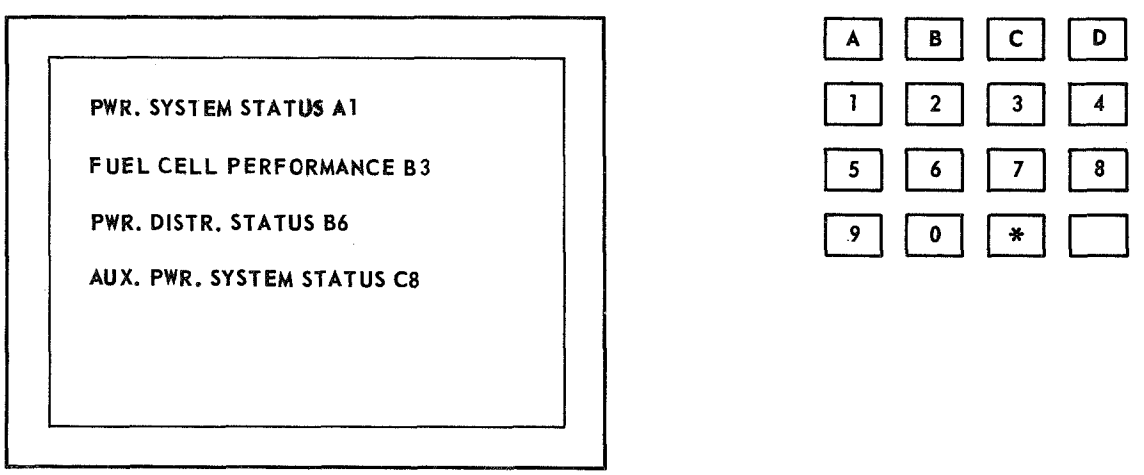
For the shuttle control, a mixture of programmable pushbuttons and keyboards appears to be the most desirable. A trade off to determine the proper mixture will be necessary to optimize space and operator convenience.



(A) LABEL ON PUSHBUTTONS



(B) LABEL ADJACENT TO PUSHBUTTON



(C) CRT KEYBOARD CUEING

Figure 7-11. Reprogrammable switch techniques.

7.4.8 Manual Flight Controls

The available technology which is adopted for this design consists of a yoke, rudder pedals, and pedestal-mounted jet engine throttle for the aircraft mode and attitude and translation hand controllers for the spacecraft mode.

A desirable objective would be to combine the yoke and pedals with the attitude controller, thus eliminating the need for separate controls. Programs are presently underway to evaluate the use of an attitude hand controller in an aircraft. This evaluation is being conducted in dynamic crew station simulation evaluations and the F-4 fly-by-wire program. Performance, pilot acceptance, and complexity/reliability of such a system will determine its final disposition.

Twist knobs and thumb wheels are still the most desirable for functions such as communication equipment operation, cabin environment control, etc. Touch tuning has been demonstrated by Collins Radio Company. A system of this type might be used to operate the communication equipment, if proven feasible.

7.5 Data Bus Interface

Two interfaces between the control and displays system and other vehicle systems are shown in the functional block diagram, Figure 7-2. Information flow across these two interfaces is dependent on the number of variables and data rates. Data rate, as referred to here, is the updating rate of a variable. Information flow from the control and display system is directly to the computer and not via the data bus. These data will consist of interrogations and commands from manual controls; therefore, the rate is negligible as far as sizing the data bus is concerned. Data to the C&D system will be fed from both the data bus and the computer. Practically all of the data available on the data bus or generated by the computer may be required by the display system at some time. The exact quantity cannot be established at this time and will depend on the individual system display and checkout requirements. The number of functions required at any one time will be relatively low compared to total information available. Therefore, the determining factor in ascertaining interface requirements is the data rate.

7.5.1 Display Update

The determinant of required data rate is the natural frequency of the displayed variable. In general, the update rate should be at least double

the natural frequency of the displayed variable or double the response rate of the observer, whichever is lower. The average visual response rate of an individual is approximately 4 Hertz, thereby establishing an upper limit for update requirements for observer comprehension at 8 Hertz. The lower limit is established by the rate of change of the variable itself. However, the primary consideration is to prevent jumpiness in the display. This is a visual problem and has nothing to do with the bandwidth of the observer. Experience with VSD in the aircraft industry has shown the update requirements for these devices range from 0.5 to 100 hertz.

Typical rates for an aircraft EADI are the following:

Roll	100 Hertz
Heading	2 to 3 Hertz
Altitude	2 to 3 Hertz
Pitch	2 to 3 Hertz
Airspeed	25 Hertz

HSD data rates are so slow they are not considered a problem. The only exception to this is when an HSD is operated in another mode; then the same rates as for a VSD apply. MFD update rates do not exceed those of the EADI.

7.5.2 Field Rate

The required field rate is derived solely from the characteristics of the eye. The design goal is a display free of flicker. The flicker threshold is dependent on the viewing-distance-to-screen-diameter ratio, the field rate, and the phosphor characteristics. Studies conducted by various people [7,8, and 9] normally show the critical flicker frequency (CFF) to reach an asymptote around 60 Hertz. These figures do not consider conditions where the background is of high intensity levels and images on the display consist of narrow lines. Under conditions of this type the CFF tends to go much higher. Manufacturers of present day state-of-the-art displays utilize frame rates in the order of 50 Hertz to overcome flicker. Memory capability is included in the display system to accommodate this frame rate and the refresh rate of 100 Hertz. By this technique the data rate demand on the data bus is limited to the rate established by the update requirements.

8.0 Data Management

8.1 Introduction

This section evaluates the subsystem requirements and establishes a conceptual data management subsystem. The data management subsystem is composed of the data distribution system, the computer system, and their interfaces. Table 8-1 is a summary of the physical characteristics of the data management subsystem.

The data distribution system consists of 1 master data processor, 3 data processors, and 37 data distribution units. As a result of data source distribution, the data processors are physically located in the front, in the middle, and in the rear of the orbiter. The 37 data distribution units are remotely located throughout the vehicle as required. Estimated total weight of the data distribution system is 29 pounds; the volume is 0.27 cubic foot. A power consumption of 28 watts is required for the data distribution system.

The computer system consists of a multiprocessor-type configuration and a small uniprocessor (4K memory) dedicated to a strapdown IMU. The multiprocessor has three processors, two input/output units, 12 modules of 8K main memory units, and a bulk memory unit (magnetic tape). The computer system is estimated to weigh 55 pounds, requires 240 watts, and has a volume of 1.2 cubic feet.

The entire data management system weighs 84 pounds, consumes 268 watts of power, and occupies 1.47 cubic feet of volume.

8.2 Data Distribution System

A mutual exchange of needed data from one point to another requires a common denominator for all of the avionics, namely, a "data bus," and a means of combining this data in an orderly manner. The basic means of accomplishing this task is referred to as multiplexing. The Data Distribution System (DDS) is defined as the means by which communications between IAS subsystems is accomplished, via the data bus. A logically designed DDS will help achieve the following goals:

a. Autonomy

b. Flexibility

TABLE 8-1. PHYSICAL CHARACTERISTICS OF DATA MANAGEMENT SUBSYSTEM

	Quantity	Weight (lb)	Power (W)	Volume (cu. ft.)
Computer System				
8K Memory Unit	12	30	120	0.6
Processor Unit	3	10	60	0.3
Input/Output Unit	2	5	20	0.1
Bulk Memory Unit	1	5	20	0.1
Strapdown Uniprocessor	1	5	20	0.1
Subtotal		55	240	1.2
Data Distribution System				
Master Data Processor	1	4	3	0.06
Data Processor	3	6	6	0.04
Data Distribution Unit	37	19	19	0.17
Subtotal		29	28	0.27
Total		84	268	1.47

c. Reliability

d. Volume reduction

e. Weight reduction.

To meet these goals, the avionics on board the spacecraft must be integrated. This appears to be the best way of achieving automatic check-out and configuration flexibility. To fully achieve our goals, the data have to be made available at a common point or on a common bus and a means of extracting the data has to be devised. The DDS consists of a data bus and necessary electronics for getting the data on or off the data bus. In this report, the data bus is the transmission medium for the data. To integrate the avionics automatically means to multiplex the data. To have all the information present when needed requires a means of coding and decoding the data. The type of multiplexing and coding is spelled out in a later paragraph of this report. The ideal DDS would allow each subsystem access to all the data all the time. This will be the goal and will be followed where practical while maintaining simplicity, reliability, and flexibility.

A DDS is by nature autonomous or self-contained from the standpoint that all data are available throughout the system. Its use makes self-checkout of IAS subsystems possible, which adds to the overall system autonomy. A minimum of ground support equipment will be required for prelaunch checkout. Flexibility is readily achievable, since the system is computer controlled and can be programmed for the desired controls and sequences. All hardware performing the same function will be standard hardware. The word standard means the hardware will be fabricated to the same documentation, and therefore, will be mechanically and electrically interchangeable. Since this will eliminate multiple design of the same function, overall cost should be reduced. Since there are fewer types of hardware, maintenance should be simpler and the percentage of spares will be reduced.

Reliability is a subject that requires more analysis. The degree of reliability that can be achieved will determine to a large extent the degree to which the avionics will be integrated. The means of achieving reliability can be accomplished in many ways. Large-scale integration (LSI) can be used in the design and development of the DDS hardware. LSI has proven to be more reliable than discrete components. Another area of reliability improvement is the reduction in connectors by utilizing the data bus. Connectors have historically created reliability problems. A reduction of connectors is feasible through elimination of terminal boxes that are presently used in a hard wire design.

The data bus concept allows the elimination of large cable bundles, improved electronic packaging, and the reduction of duplicated efforts, which in turn reduces the volume and weight required for the spacecraft.

To better explain the data bus concept, a comparison will be made with a simple system. A hard wire approach is shown in Figure 8-1. The same system is shown in Figure 8-2, utilizing a data distribution technique. This comparison illustrates the elimination of large bundles of wire and terminal boxes by utilization of the data bus and associated electronics.

Figure 8-3 illustrates a preliminary shuttle DDS, oriented toward providing subsystem automony. The orbiter was chosen to illustrate the data bus concept since it should result in worst-case condition because the orbiter has to perform all of the functions the booster does, plus several more. Even though the booster has more engines the orbiter has greater data requirements to meet other mission phases, such as rendezvous and docking. The same basic concept will apply to the booster data distribution system.

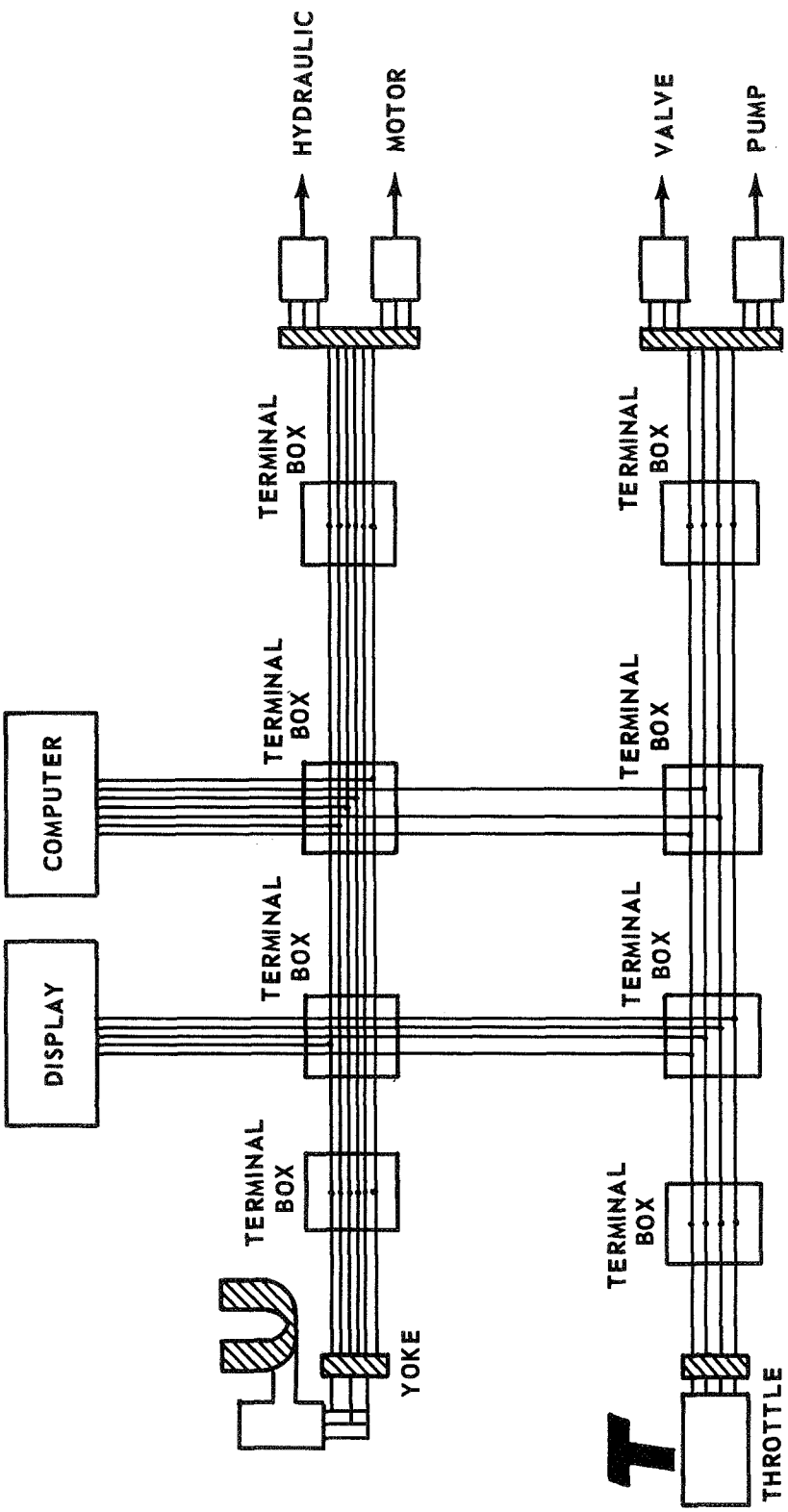


Figure 8-1. Hardwire approach.

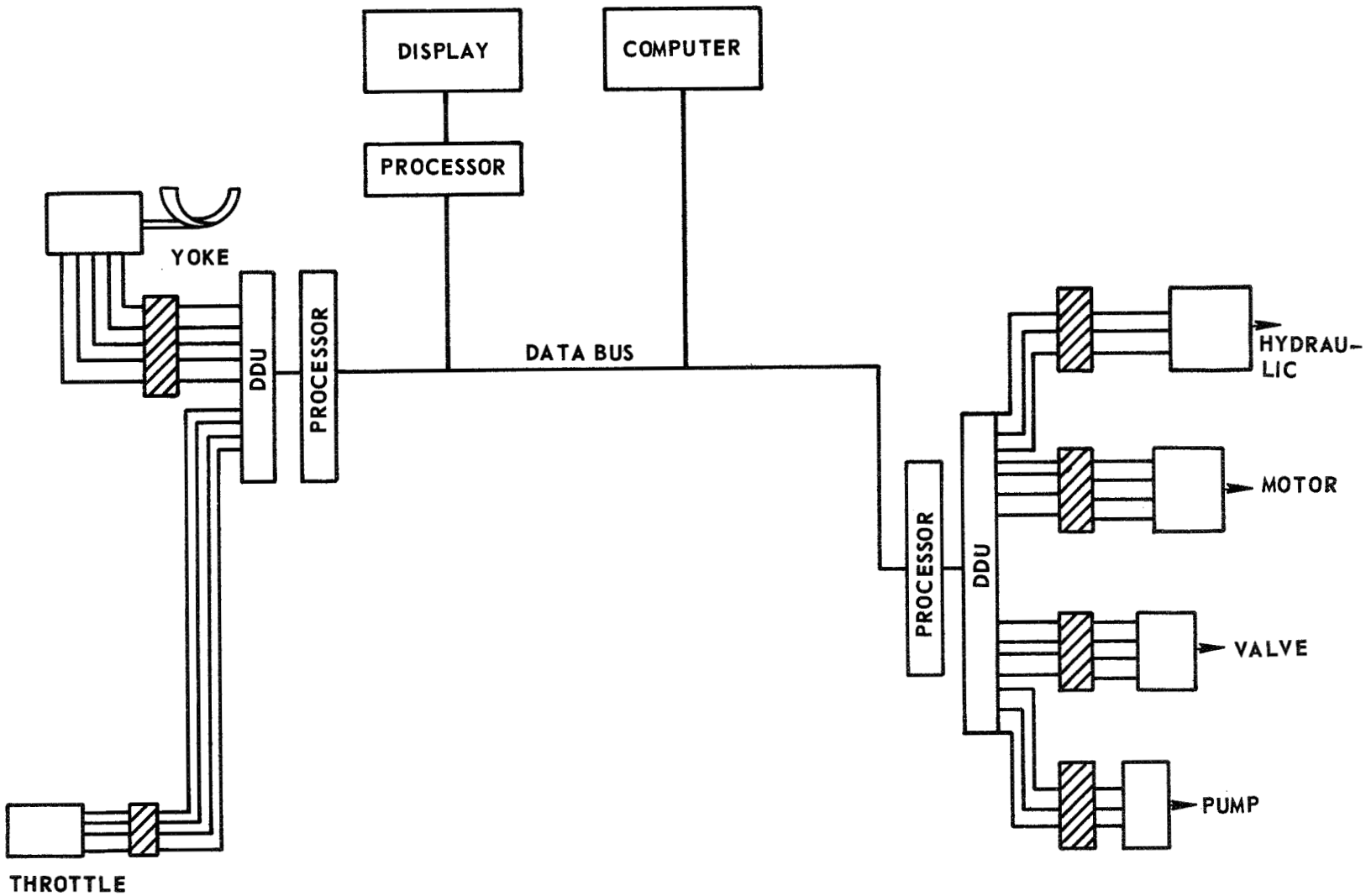


Figure 8-2. Data bus technique.

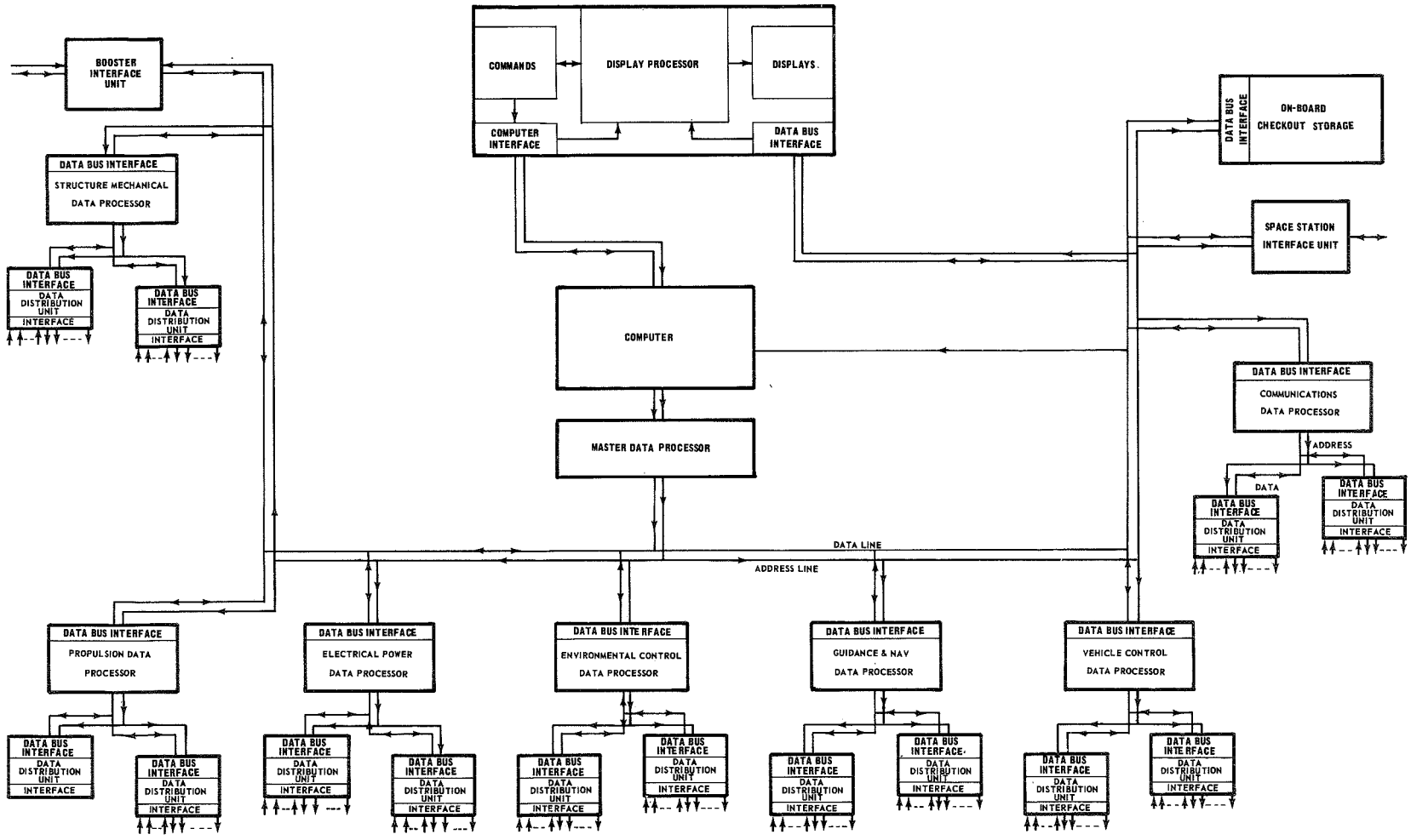


Figure 8-3. Preliminary shuttle data distribution system (subsystem autonomy).

The term data bus refers to the two pairs of wires (twisted shielded pair), shown in Figure 8-3, that are connected to all subsystem data processors. These two pairs of wire (4 wires) will replace the bundle of wires in a conventional space vehicle. One pair is used to transmit and receive the data; the other pair is used to transmit the address of the data.

8.2.1 Requirements

The general requirements for the shuttle IAS were given in Section II. The requirements iterated in this section should be considered as supplementary details to those general requirements. Naturally, the general requirements have a major effect on the data management concept.

8.2.1.1 Data Requirements

The data used in design and analysis of the data distribution system are presented in Appendix A. The following guidelines were used to correlate the data:

- Data points are assigned to their source.
- Power distribution and control are not included.
- Worst case estimates of the number of data points.

System sampling rates are graphically illustrated in Figure 8-4. Average system sampling rates are 3.2 samples/sec for test points and 7.0 samples/sec for interfaces and intrafaces. These averages are based on a minimum sampling rate of 1 sample/sec and a maximum of 50 samples/sec. A sampling rate of 50 samples/sec is assumed for all control data points which is considered to be a conservative worst-case condition.

A significant fact apparent from the data requirements is that 35 percent of the data points are discretes. This may be used to reduce the bit rates by transferring 10 discretes/word. The data flow diagram is shown in Figure 8-5 and illustrates the number and relationship between control points (167), interfaces (265), test points (975), and intrafaces (528). The total number of data points is 1935. This number was used in developing the DDS concept.

The data distribution in one axis is shown in Figure 8-6. This illustrates the physical location of the data points which will govern the design of the data distribution system.

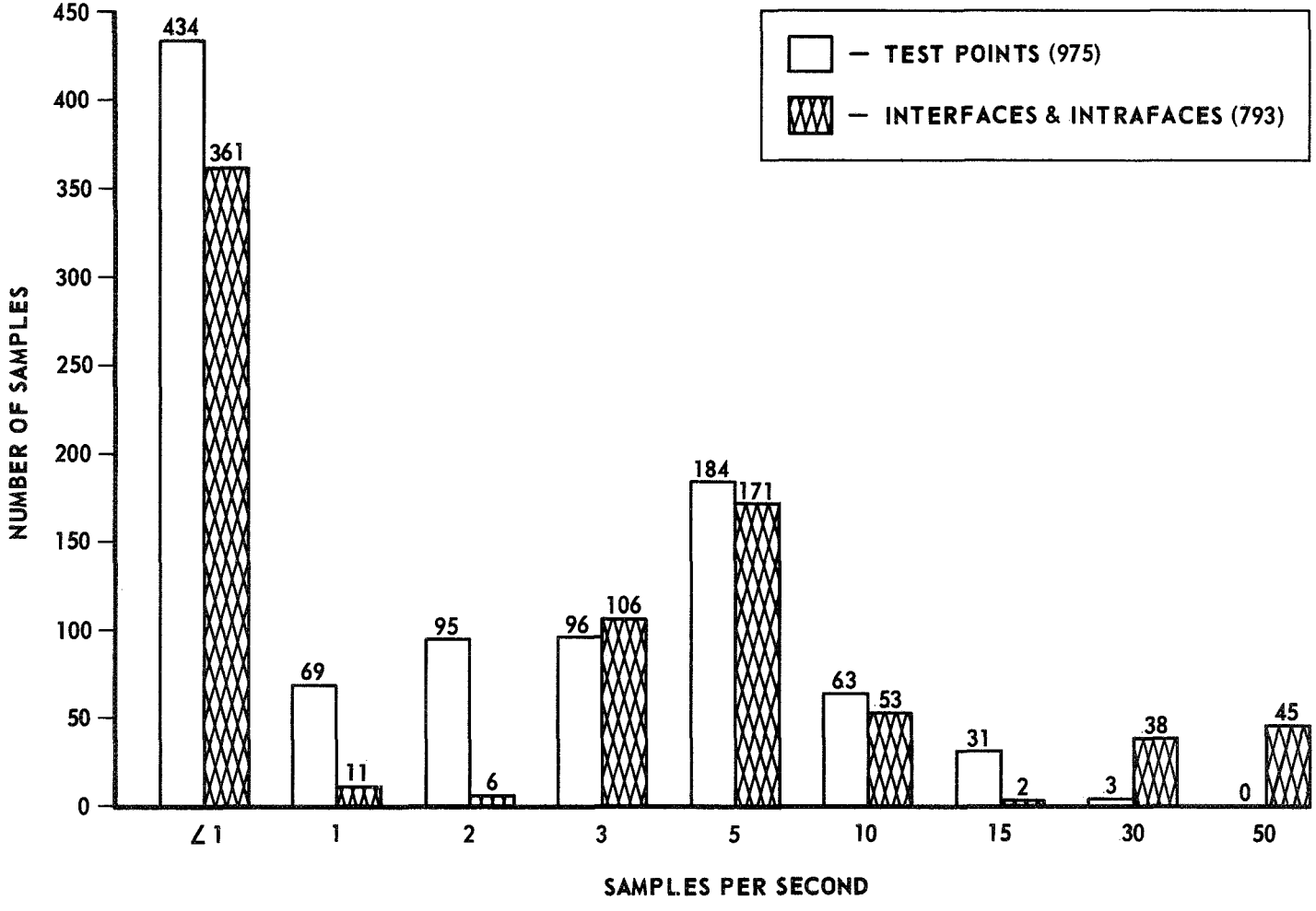


Figure 8-4. System sampling rates.

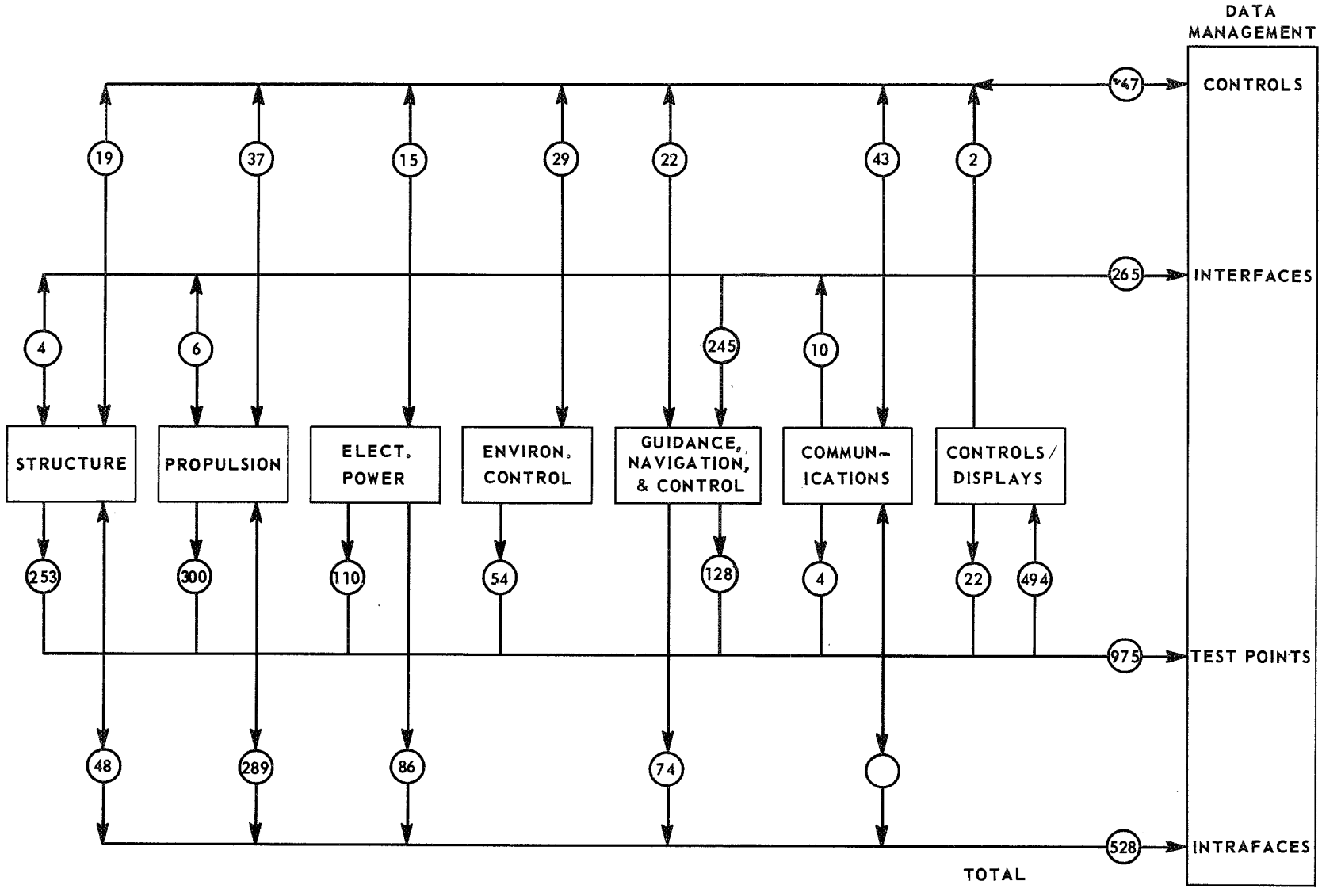


Figure 8-5. Data flow summary.

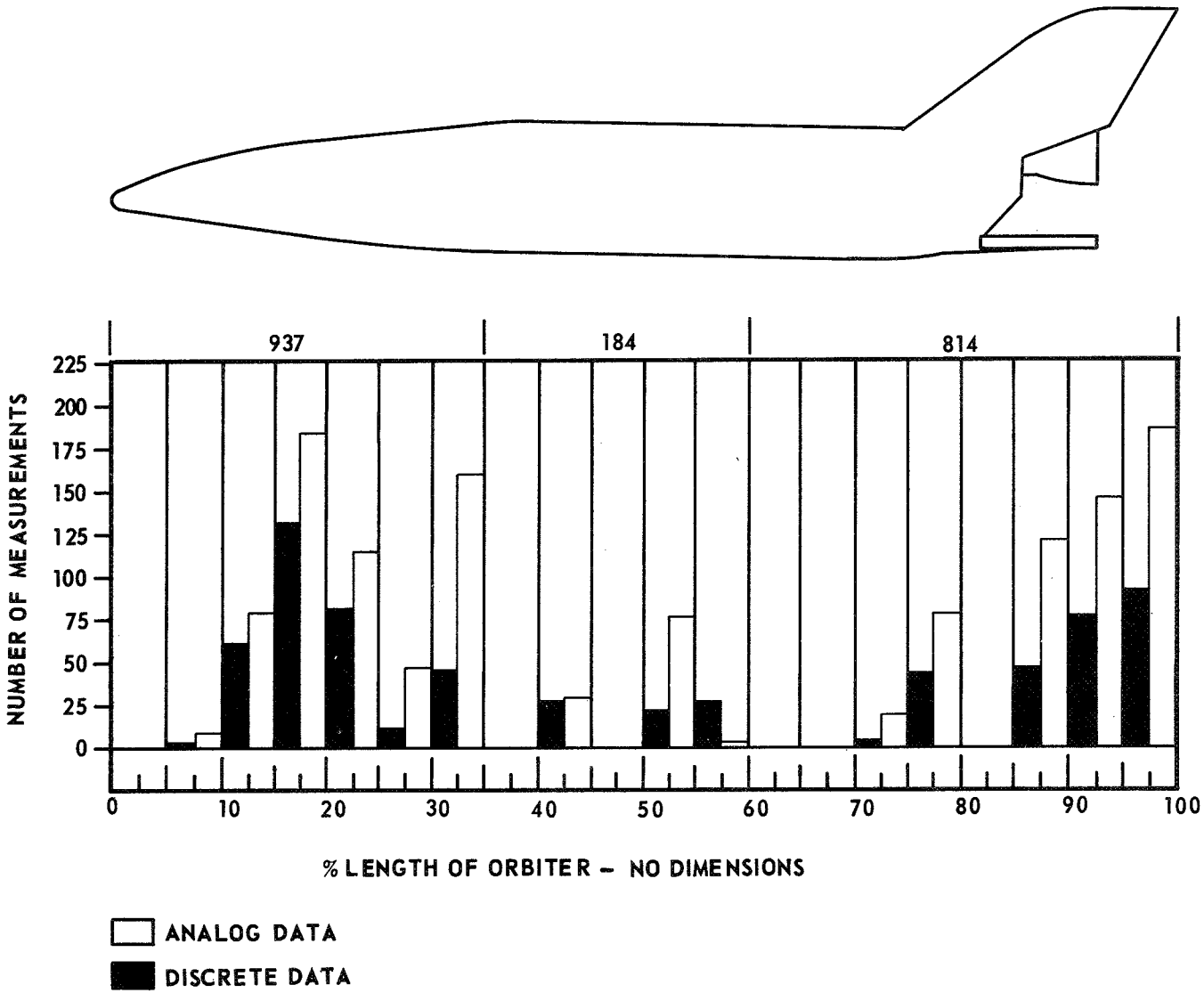


Figure 8-6. Data distribution in one axis.

8.2.1.2 Hardware Distribution Requirements

The hardware distribution is shown in Figure 8-7. The hardware distribution is required to determine the physical routing of the data bus. Figure 8-7 identifies hardware according to the subsystem discipline. For example, it illustrates that the inertial measurement unit belongs to the guidance, navigation, and control subsystem. Subsystems are shown horizontally and placement of hardware from nose to tail is shown left to right.

8.2.1.3 Preliminary Design Concept

An initial orbiter DDS was selected considering the eight different shuttle subsystems. This is shown in Figure 8-3. This system was then iterated until the final system was determined. This DDS approach allows each subsystem to be autonomous or perform the internal control and sequencing that is required by that individual subsystem. However, data required from other subsystems has to be acquired via the data bus. Subsystem autonomy reduces the main data bus bit rates by allowing the subsystem to operate at its own natural frequency. In this concept, if a failure occurred in DDS, such as the data processor or a data distribution unit, it would only affect that particular subsystem where the failure occurred. This initial concept is based on the idea that each subsystem has its own data processor and required data distribution units.

After considering the hardware distribution requirements, as shown in Figure 8-7, it became apparent that the initial approach was not the optimum data bus concept. Since the hardware for any one subsystem is physically distributed from the front to the rear of the vehicle, multiple data buses are required if the independent subsystem design is maintained. As many as five subsystems would require wire routed the full length of the vehicle; obviously, this does not minimize cabling. One solution to this situation is to consider the data distribution as a function of the hardware location and design the DDS accordingly (Fig. 8-6). The data as shown in Figure 8-6 are conveniently grouped into three main areas of the vehicle; the front, the middle, and the rear. A total of 937 data points are in the front; 184 are in the middle and 814 data points are in the rear of the orbiter. Having the three basic data groups controlled by the three data processors results in the vehicle DDS shown in Figure 8-8.

As indicated in Figure 8-8, all stations that connect to the main data bus or sub-data bus are connected by a standard data bus interface. The data processors and data distribution units are standard and may be added or removed as required. This flexibility is highly desirable since a measurement list, such

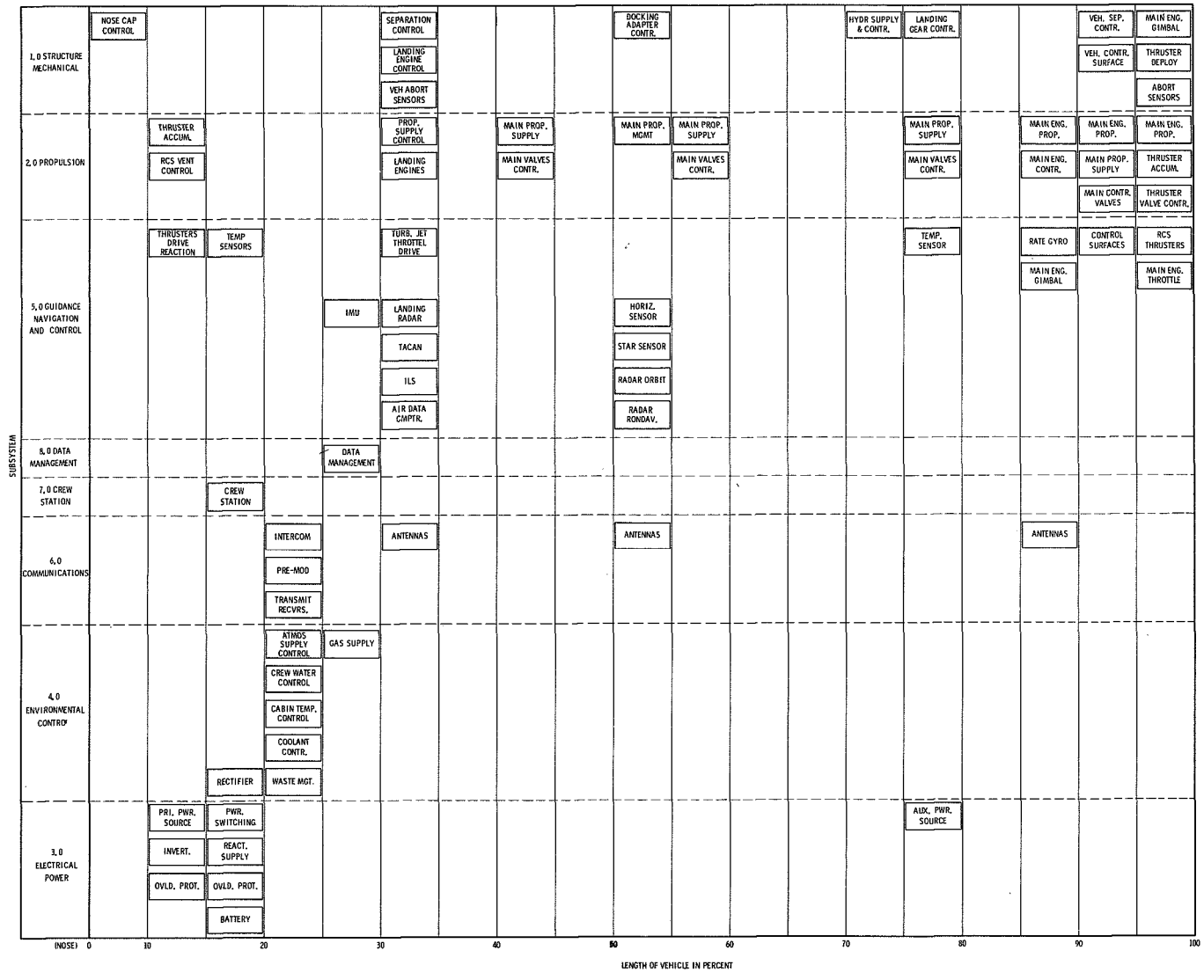


Figure 8-7. Orbiter hardware distribution.

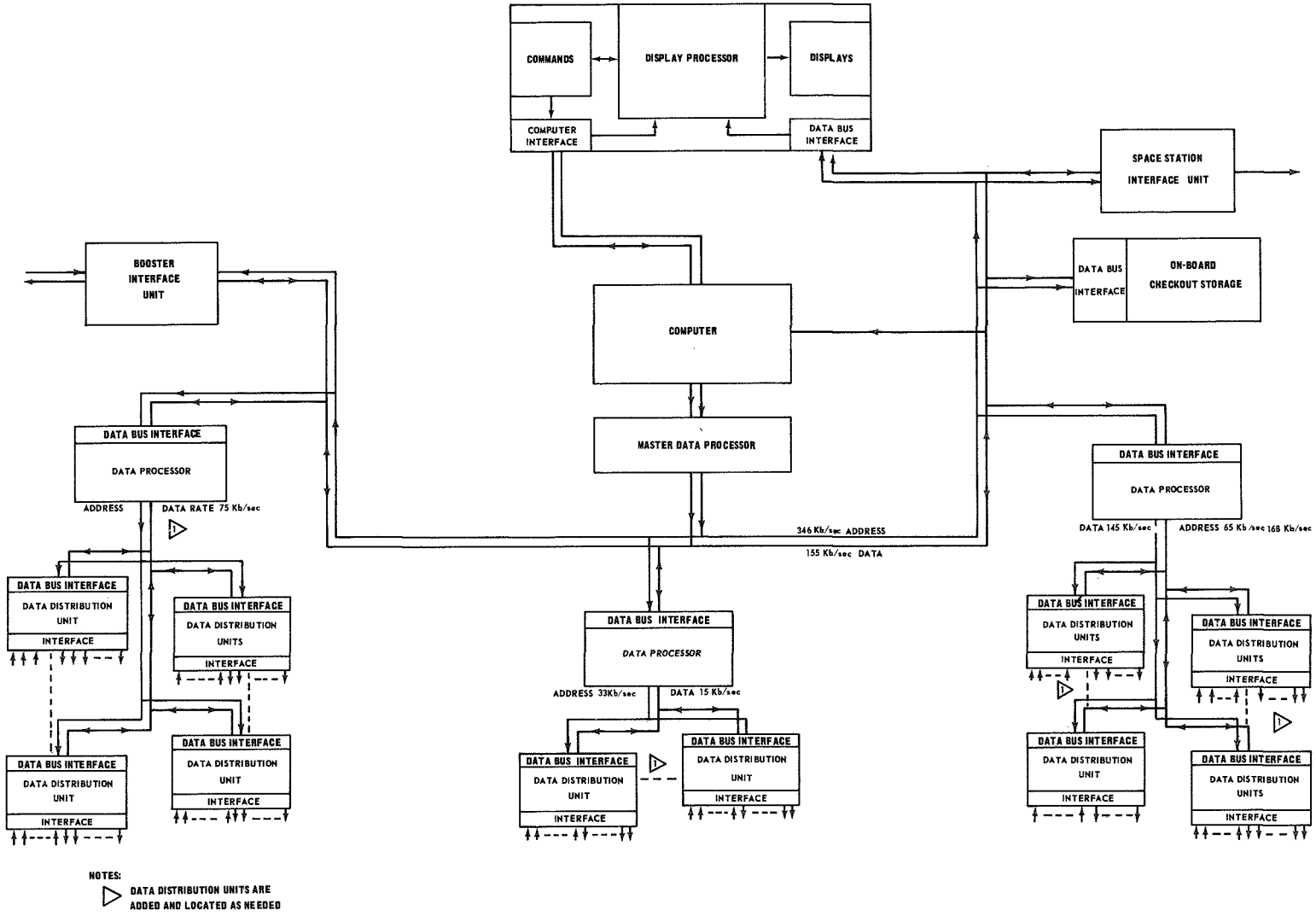


Figure 8-8. Orbiter data distribution system conceptual design.

as indicated in Appendix A, is constantly changing as the program develops. The data requirements will change during the hardware design phase of the program as well as in development and operation. These changes can be made without a DDS hardware redesign. The data bus design may be broken into the following categories for detailed comments:

- Computer
- Master data processor
- Transmission media and access techniques
- Modulation techniques
- Data bus interface
- Data processor
- Data distribution unit
- Data bus bit rates

8. 2. 2. 1 Computer

A detailed computer analysis for the shuttle is given in Section 8. 3. Functions of the computer in relation to the data bus are listed below:

- Primary control of the data bus.
- Interfaces with the crew for command and display of any measurement via data bus.
- Monitors the data bus and accepts data required by the computer.
- Recognizes emergency conditions, and exercises interrupt command into the master data processor.
- Updates the memory in the master data processor according to the mission phases.
- Performs limit checking and address generation for on-board checkout.

8. 2. 2. 2 Master Data Processor

The master data processor performs the routine data bus control and sequencing, thus allowing the computer to perform more complex analysis. Primary duties of the master data processor are the following:

- Controls data flow on the main data bus.
- Transmits commands to data processors for switching modes with change of mission phases.
- Capable of being updated by the computer.
- Transfers data from one subsystem to any other subsystem via the data bus.

A functional diagram of the master data processor is illustrated in Figure 8-9. When an interrupt command is initiated, the computer gains control of the data bus and injects the desired addresses as long as the interrupt command is applied. Once the interrupt command is removed, the master processor continues its normal function.

8. 2. 2. 3 Transmission Media and Access Techniques

Various transmission means were explored for the data bus. Some were eliminated early in the program because of difficulty of applying the technique to the data bus concept. Two such schemes were fiber optics and laser techniques. The broad frequency data spectrum these afford is not required. The search was narrowed to two existing applicable transmission lines, coaxial cable and twisted shielded pair cable. These two are compared in Table 8-2.

The ease of installation and the ability to apply a differential input/output connection to the hardware makes the twisted shielded pair the preferred transmission line. Ease of installation relates to the simplicity of soldering connectors and the flexibility of physically installing the transmission line in the vehicle. As indicated, the twisted shielded pair has a high frequency cutoff of approximately 1 megahertz without utilizing highly specialized designs. This upper frequency cutoff is not a limitation since the maximum data rate is only 50 percent of the 1 megahertz capability.

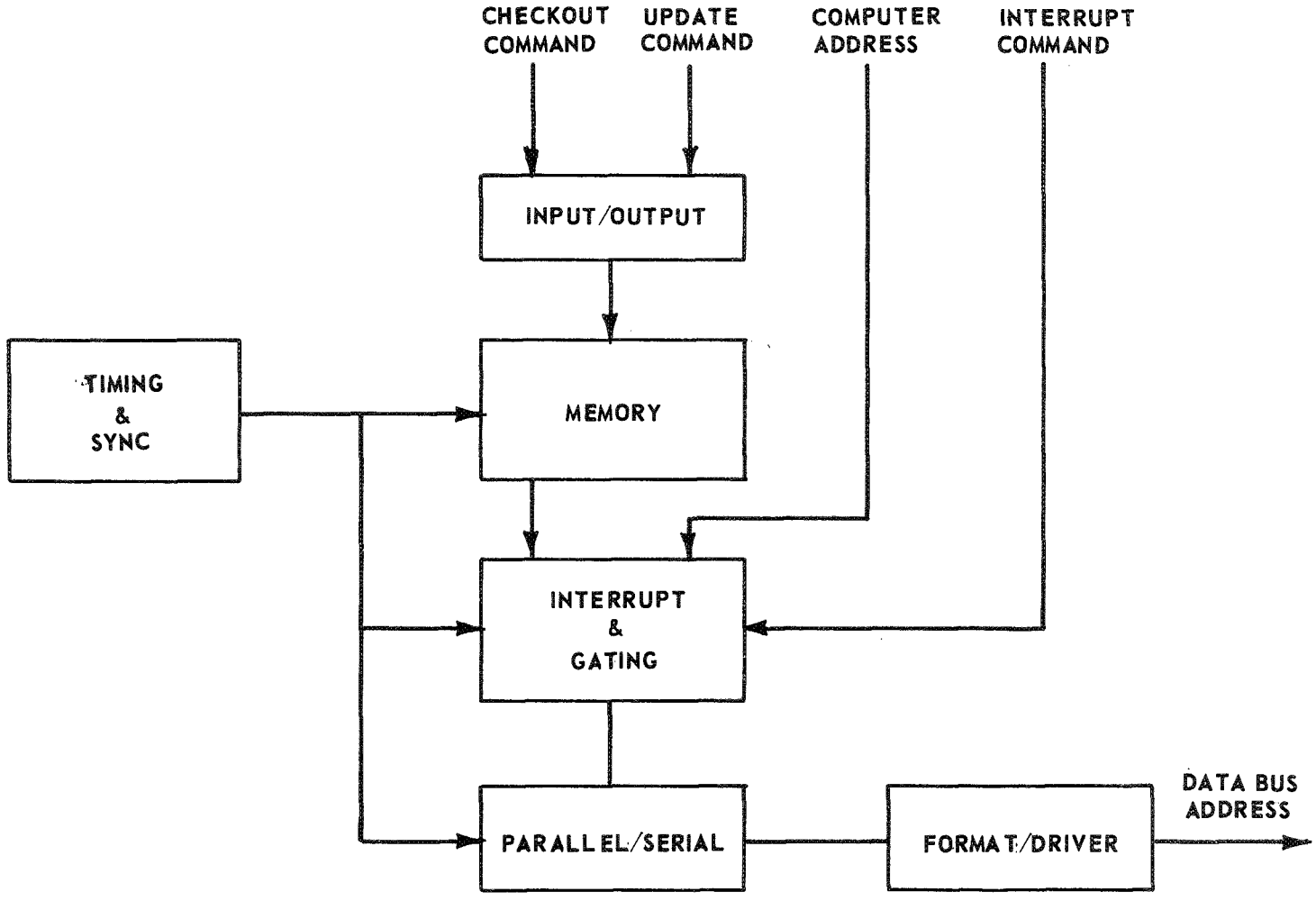


Figure 8-9. Functional block diagram of the master data processor.

TABLE 8-2. A COMPARISON OF TRANSMISSION LINES

	Frequency Response	Ease of Installation	Termination	EMI
Twisted shielded pair	Up to 1 megahertz without significant problem	Easy to work with	May use differential or single ended	Acceptable
Standard coaxial cable	In the gigahertz range	Much harder to work with	Single ended only	Acceptable

Two techniques for gaining access to the data bus were investigated. Figure 8-10 depicts a transformer approach to accessing on and off the data bus while Figure 8-11 shows a differential amplifier design. Laboratory results indicate that both designs are acceptable. Both approaches account for a differential design which reduces the noise level because of common mode rejection. A single-ended concept does not offer common mode rejection. The transformer has the advantage of acting as a bandpass filter when the transformer is optimized around the system basic frequency. This adds interference reduction capabilities to the system. In addition, the center-tapped transformer design provides added isolation plus a degree of open and short circuit protection. The degree of protection is based on the fact that to short or open either side of the transformer has the effect of changing the turns ratio. The impedance that is reflected is proportional to the square of the turns ratio. From this it can be seen that a short or open will not adversely affect the operation of the data bus. These characteristics led to choosing the transformer technique for the conceptual design.

8.2.2.4 Modulation Techniques

Time-division multiplexing was chosen as a baseline for the data bus. Multiplexing can be divided into two types: frequency-division multiplexing and time-division multiplexing. Frequency-division multiplexing is used to transfer high frequency data, while time-division multiplexing is used to transfer low frequency data. When using frequency-division multiplexing, the data are transformed into the frequency domain. The number of frequencies required will be dependent on the number of data points. When using time division multiplexing, the data are transformed into the time domain. The amount of equipment

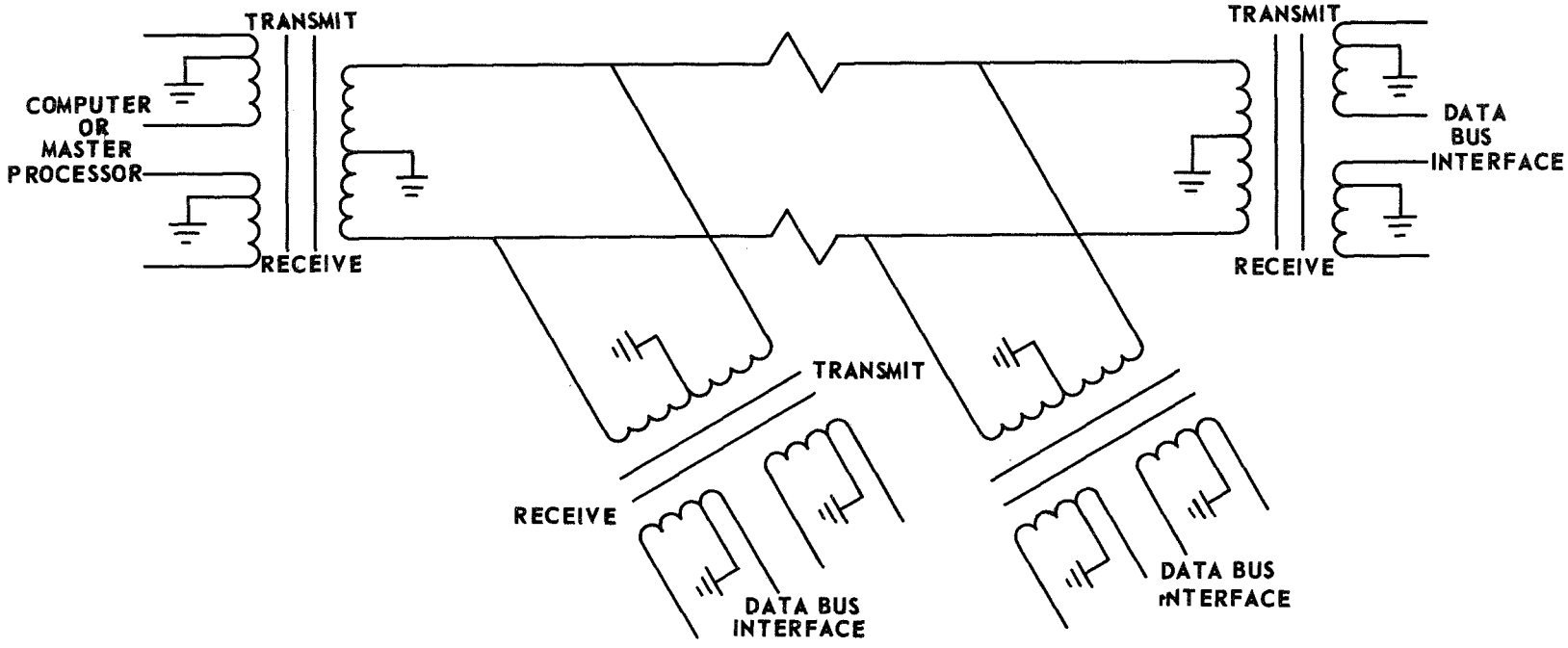


Figure 8-10. Access on/off data bus using transformer technique.

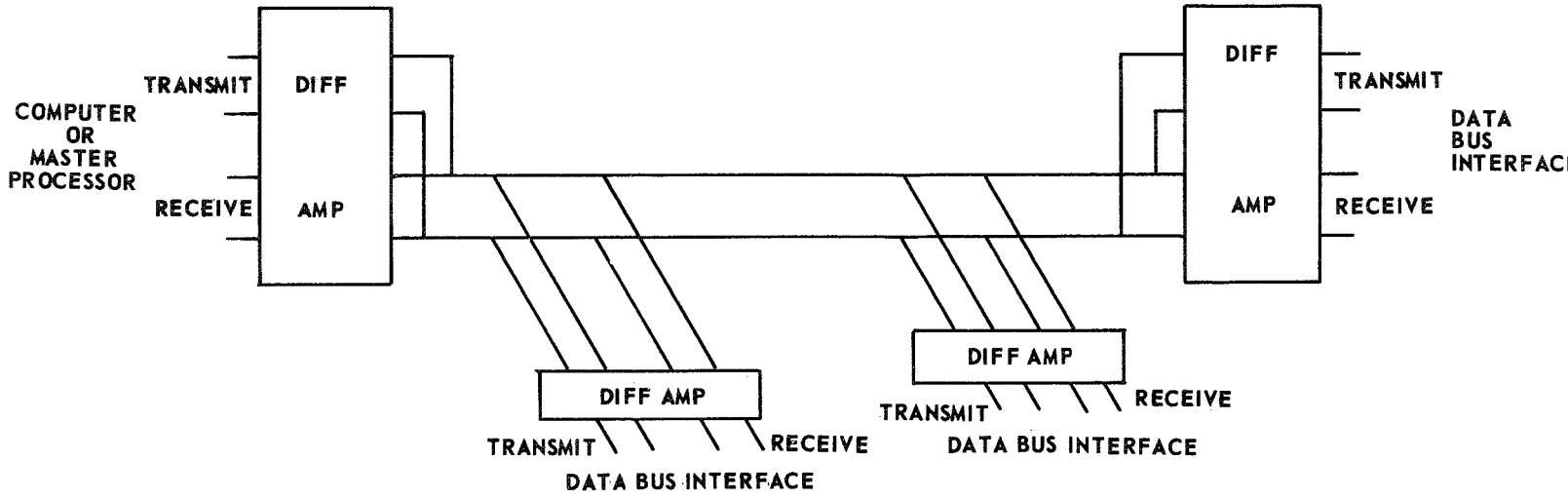


Figure 8-11. Access on/off data bus using differential amplifier technique.

required for frequency-division multiplexing is greater than that required for time-division multiplexing. The interface for frequency-division multiplexing is more complex because of the different frequencies required to transfer data. Time-division multiplexing can be handled by a common interface; the only added requirement peculiar to this method of data transfer is that data have to be coded.

Pulse amplitude, pulse duration, and pulse code modulation (PCM) were considered for the time-division multiplexing. Pulse amplitude and pulse duration modulation are basically analog methods which have inherent noise, stability, distortion, and linearity problems. PCM avoids these normal analog problems, due to its binary structure. In addition, PCM leads to fewer problems when interfacing with the digital computer since both utilize digital techniques.

Various PCM waveforms were considered such as nonreturn to zero (NRZ), return to zero (RZ), biphase (split phase), amplitude modulation, and phase reversal keying (PRK). From this group, the last three codes were selected for more detailed analysis. These three coding schemes are shown in Figure 8-12. Biphase and PRK do not contain a dc element. All of the codes basically have the same bandwidth requirements. All three codes contain the clock frequency since every bit has a transition or a voltage change during the bit period. This allows the clock frequency to be derived from the address instead of having a separate clock line. The filtered biphase code was selected as a baseline. Among the reasons for this choice were electromagnetic interference reduction and the practical ease of generating this waveform.

8.2.2.5 Data Bus Interface

Standard data bus interfaces (DBI) should be designed and documented to eliminate multiple design and development of an interface. As indicated in Figure 8-8, the DBI is identical in all sub-buses as well as in the main data bus. A typical functional block diagram is shown in Figure 8-13. This preliminary concept evolved when the data processor, data distribution unit, onboard storage, and the displays were required to use the same data bus interface. Subsystems may be added to the data bus, if required at a later date, by utilizing the standard data bus interface.

8.2.2.6 Data Processor

The data processor is connected to the main data bus through the DBI. A processor should perform the following functions:

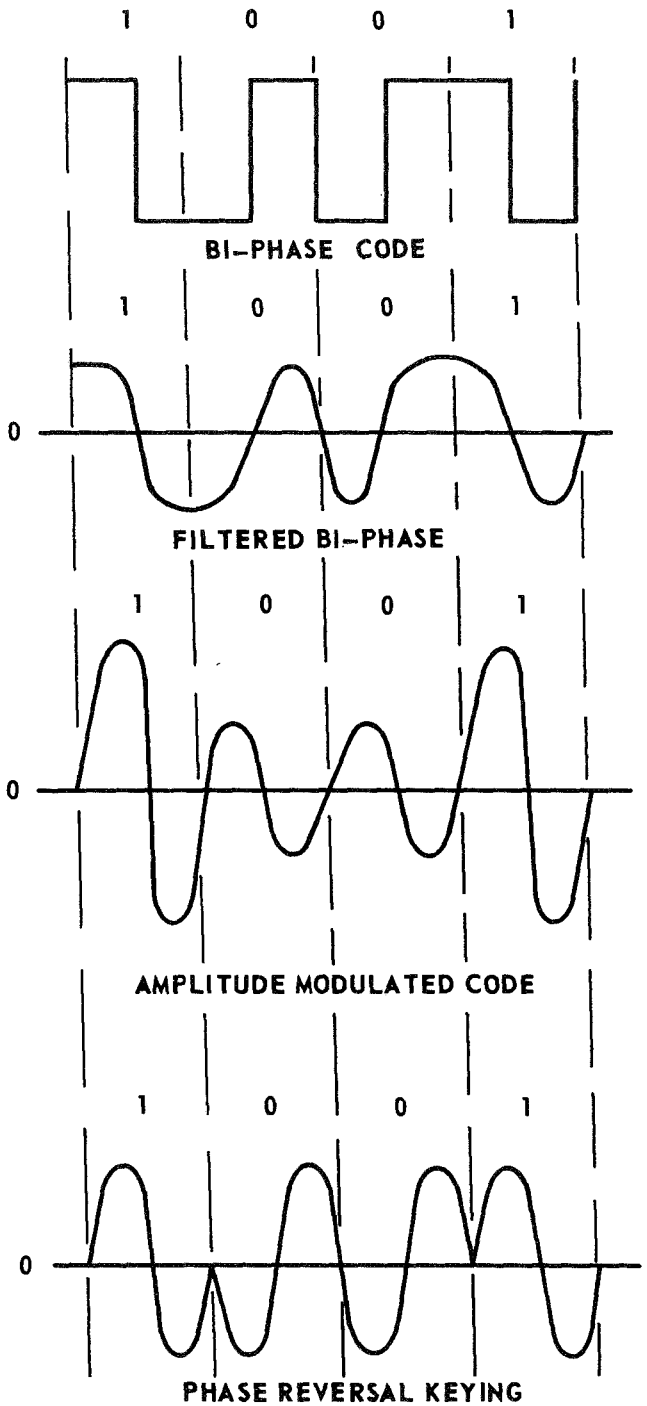


Figure 8-12. Coding techniques.

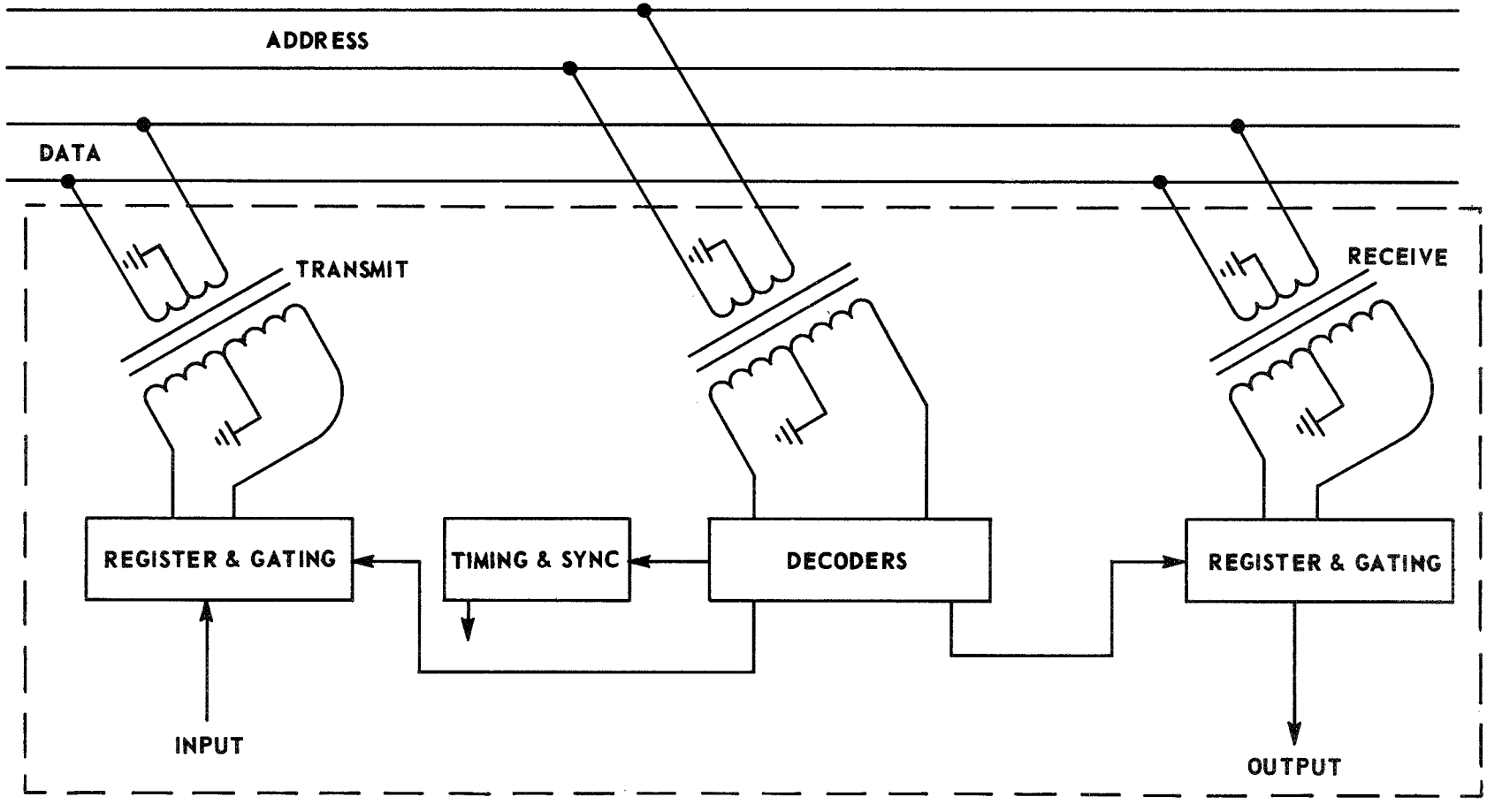


Figure 8-13. Block diagram of data bus interface.

- Address and control the data distribution units (DDU).
- Transfer data to other data processors or any main DBI.
- Accept mode commands from master processor.
- Accept checkout commands from master processor.
- Allow interrupt from computer through master processor.

The state-of-the-art is such that it is practical for one processor to handle up to 2000 data points. A block diagram of a data processor is shown in Figure 8-14. The processor will require memory to randomly address 2000 data points through the use of data distribution units. Approximately 22 percent of the data under control of one data processor is routed from one distribution unit to another via the subdata bus without going through the data processor. This reduces the number of components handling the data and the data transfer time from one distribution unit to another. The same logic holds for transfer of data from one interface to another on the main data bus.

A processor has the memory capability to store different programs which may be selected as a function of the mission phase. These different programs change the sampling rate of any measurement as required by the associated mission phase. This is another one of the flexibility features of the DDS. Modes of the processor are selected by the master processor via the data bus.

Interrupt commands are accepted by the processor from the computer. This allows the computer to address and transfer any measurement that is required in the case of an emergency. Also the processor will accept a checkout command and sequence through a stored checkout routine. The data processor recognizes the address word, configures itself to transmit or receive and proceeds to transfer the data the address word indicated.

Data processors are obviously not as intelligent as the master processor, but have the necessary memory to perform the above indicated functions. Standardization is included in the data processors; therefore, all data processors are interchangeable. Their number can increase or decrease with negligible system impact if the data requirements change.

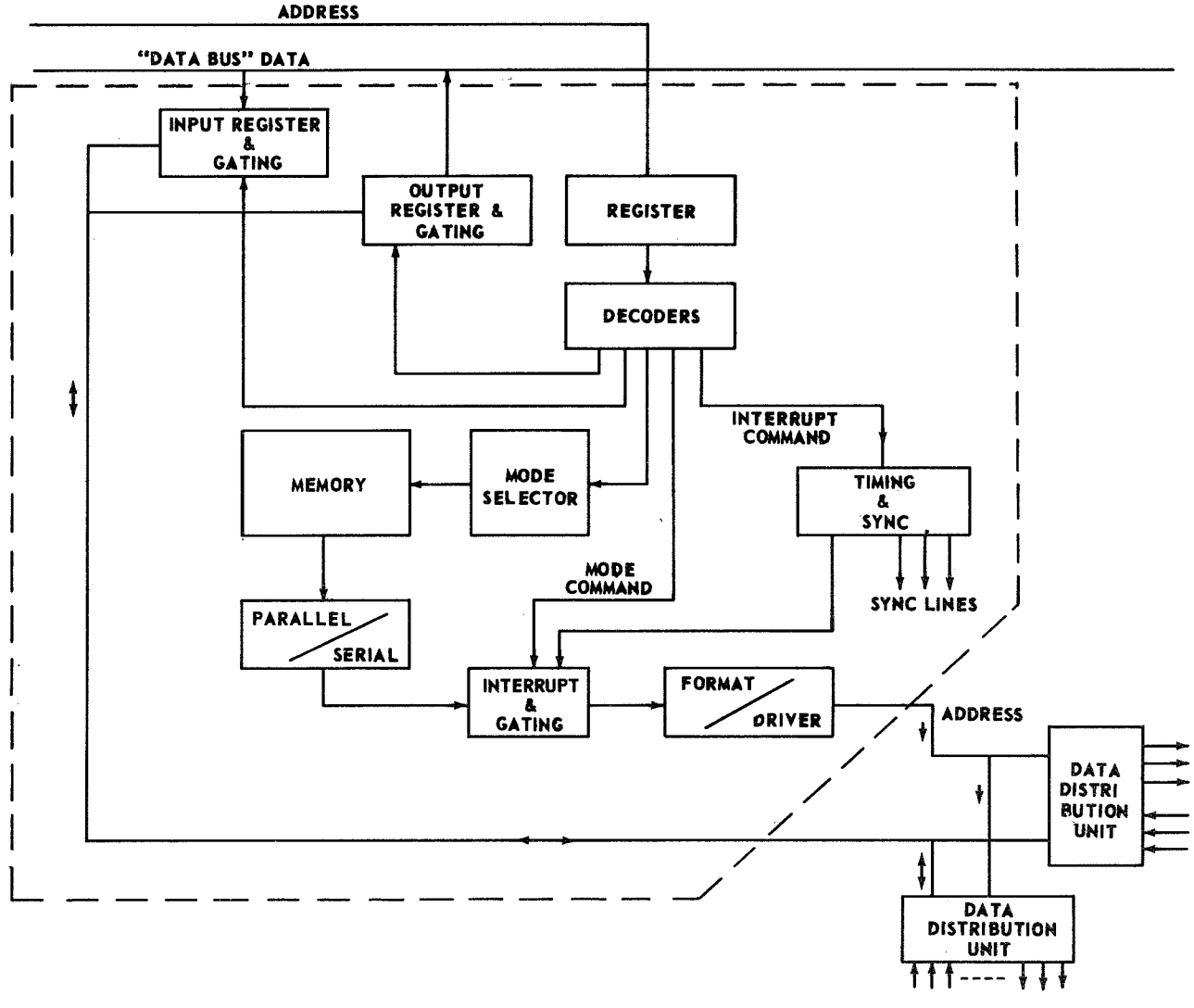


Figure 8-14. Functional block diagram of the data processor.

8. 2. 2. 7 Data Distribution Unit

Figure 8-15 functionally illustrates a data distribution unit. The data distribution unit will perform the following tasks:

- Receives address words containing instructions for transmitting and receiving data via processors and transfers the data.
- Performs analog-to-digital and digital-to-analog transformations.
- Interfaces with the various subsystems.

Basically, the distribution unit performs a gating function when addressed by the processor and performs such analog-to-digital and digital-to-analog conversions as needed. The unit will accept both digital and analog data. The ratio of numbers of analog to discrete channels will be determined from the data point list. If the input data are analog, an analog-to-digital conversion is performed at this point. This is done to provide noise immunity in the system as early as is practical. After the data are in digital form the wavetrain is converted into a filtered biphase format and accessed onto the data bus through the standard data bus interface.

The distribution unit has the capability of setting different gain levels in response to instructions included in the address word. This flexibility allows the obvious advantage of accepting different input levels from the measurement sensors. Once the address word is decoded, the data are accepted, gated through a digital to analog converter if necessary, then gated to the proper channel.

The quantity of distribution units for the orbiter is approximately 37. This is based on a 64-channel capacity distribution unit.

Figure 8-16 is a plot comparing weight of hard wired measurements versus data distribution unit technique. The graph was plotted from the following weight data:

Data distribution unit (64-channel capacity)	8 ounces
Data bus wire (2 twisted shielded #24 pair)	10 lb/1000 ft
Hard wire (#24 single conductor shielded)	5 lb/1000 ft
Reference specification	MSFC 40M39588

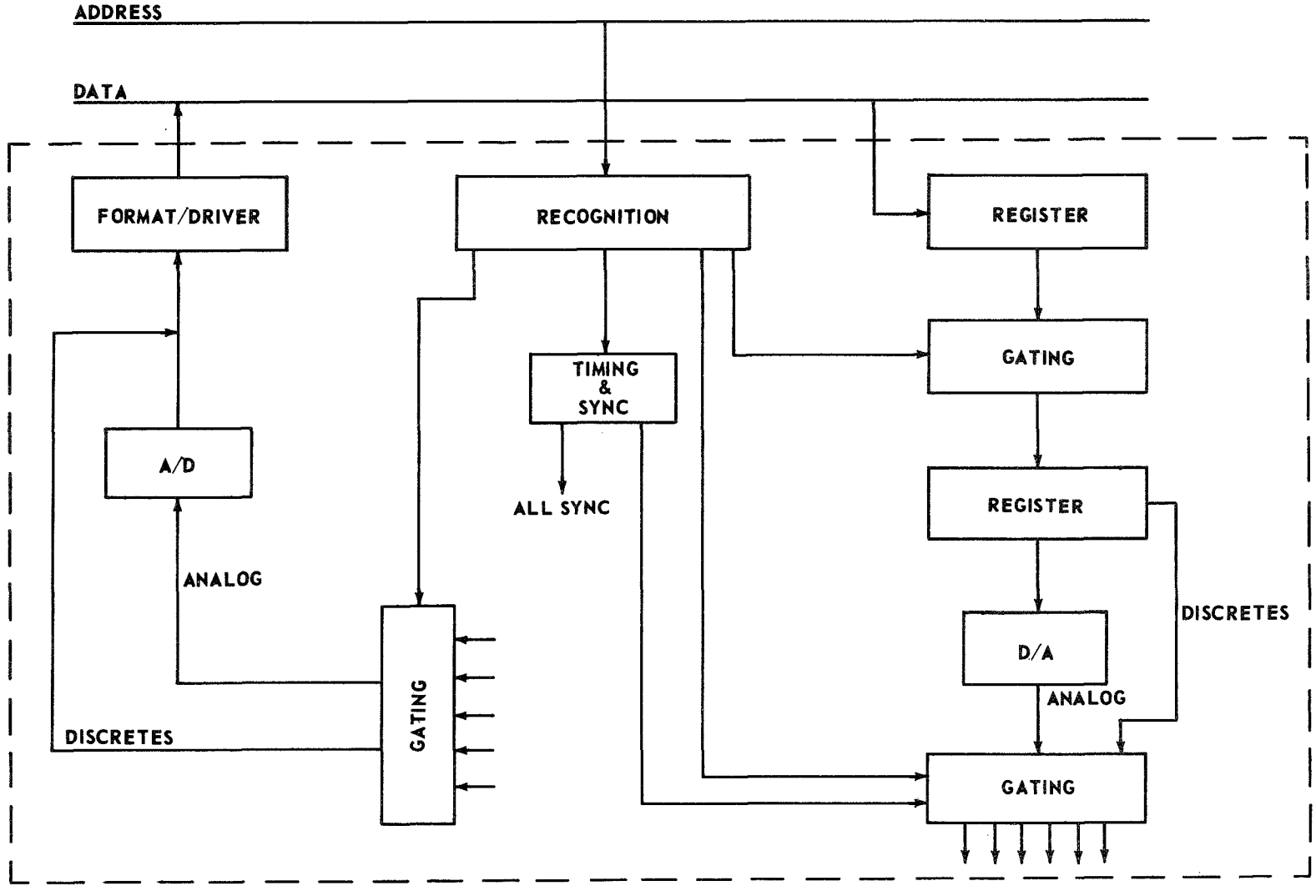


Figure 8-15. Functional block diagram of data distribution unit.

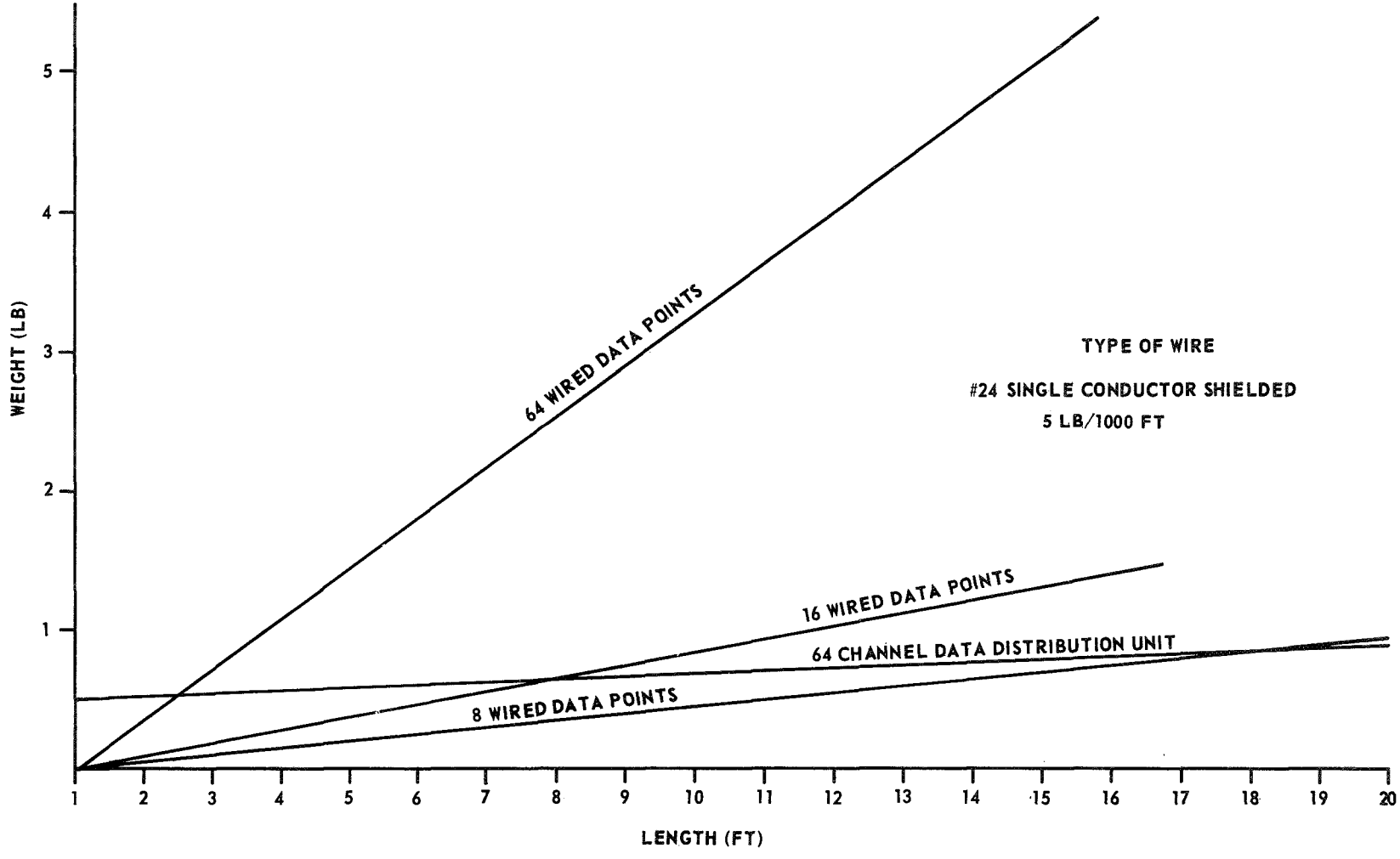


Figure 8-16. Weight comparison of hard wired versus data distribution unit.

The assumptions used were the following:

- All data points are equi-distant from the distribution unit.
- Single conductor shielded wire is used in the hard wired case with one common ground.

General guidelines concerning when to add a data distribution unit in relation to the number of data points at given distances are illustrated in Figure 8-16. This is for weight consideration only. For example, a 64-channel distribution unit weighs less than 64 hard wired data points at any distance greater than 2.5 feet. Sixteen measurements have to be routed 8 feet or greater before a 64-channel distribution unit has the weight advantage. Channel capacity may be varied by adding channels in multiples of 16 to the distribution unit. The metal oxide semiconductor (MOS) technology is presently available to implement the 16-channel design.

8.2.2.8 Data Bus Bit Rates

Bit rate calculations are based on the data requirements list in Appendix A. When considering these bit rates, it should be remembered that a worst-case condition has been assumed. The calculated bit rates are peak rates. Voice and television were not included in the bit rates for this report. These subjects are being analyzed for data bus transmission. A more detailed bit rate analysis will be performed at a later date. Figures 8-5 and 8-6 were used as guidelines in calculating these bit rates. From the data requirements section an adjusted sampling rate was determined as given in Table 8-3.

TABLE 8-3. DATA CLASSIFICATION

Type/Data	Sample/Sec	No. of Measurements	Percent of Total Data
Controls	50	167	8.6
Intrafaces + Interfaces	7	793	41
Test Points	3.2	975	50.4

Note: Adjusted sampling rate = 0.086 (50) + 0.41 (7) + 0.504 (3.2) = 9 samples/sec. This adjusted sampling rate was used to size the bit rates for preliminary consideration.

An address and a data word format may be formulated for the DDS as shown in Figure 8-17.

Addressing techniques with maximum bit utilization can be designed by programming specialists and will evolve with the hardware development; however, this simple format will suffice for a preliminary assessment of bit rates. It was found that out of a total of 1935 measurements, 677 were discrete measurements and 1258 were analog measurements. Since 10 discretes occupy only one analog measurement slot, the effective number of measurements involved is calculated as follows: 677 discretes divided by 10 added to 1258 gives an effective number of data points as 1326.

Referring to Table 8-3 for the adjusted sampling rate of 9, and the calculations above for effective number of data points as 1326, and Figure 8-17 for address format word length of 29, the main data bus address bit rate is computed as follows:

$$9 \times 1326 \times 29 = 346 \text{ Kb/sec}$$

Figure 8-14 shows that the data word format consists of 13 bits. The main data bus data rate is computed as follows:

$$9 \times 1326 \times 13 = 155 \text{ Kb/sec}$$

Forty percent of the data may require multiple address techniques. For example, two data processors receiving while another data processor is transmitting data will require these techniques. Such a condition results in address bit rates of 484 Kb/sec. Address bit rates in the order of 623 Kb/sec would be required when 40 percent of the data is transmitted from one data processor with three other data processors or main bus interfaces in the receive mode. The bit rates will be considerably less than the above numbers since these are based on all measurements being required during all phases of the mission. As many as eight distinct mission phases are now being evaluated. However, the worst-case condition only utilizes 50 percent of the design concept capabilities, thus allowing for growth.

The orbiter data distribution system concept, Figure 8-8, depicts three data processors. One data processor will be physically located in the front, one in the center, and one in the rear of the orbiter.

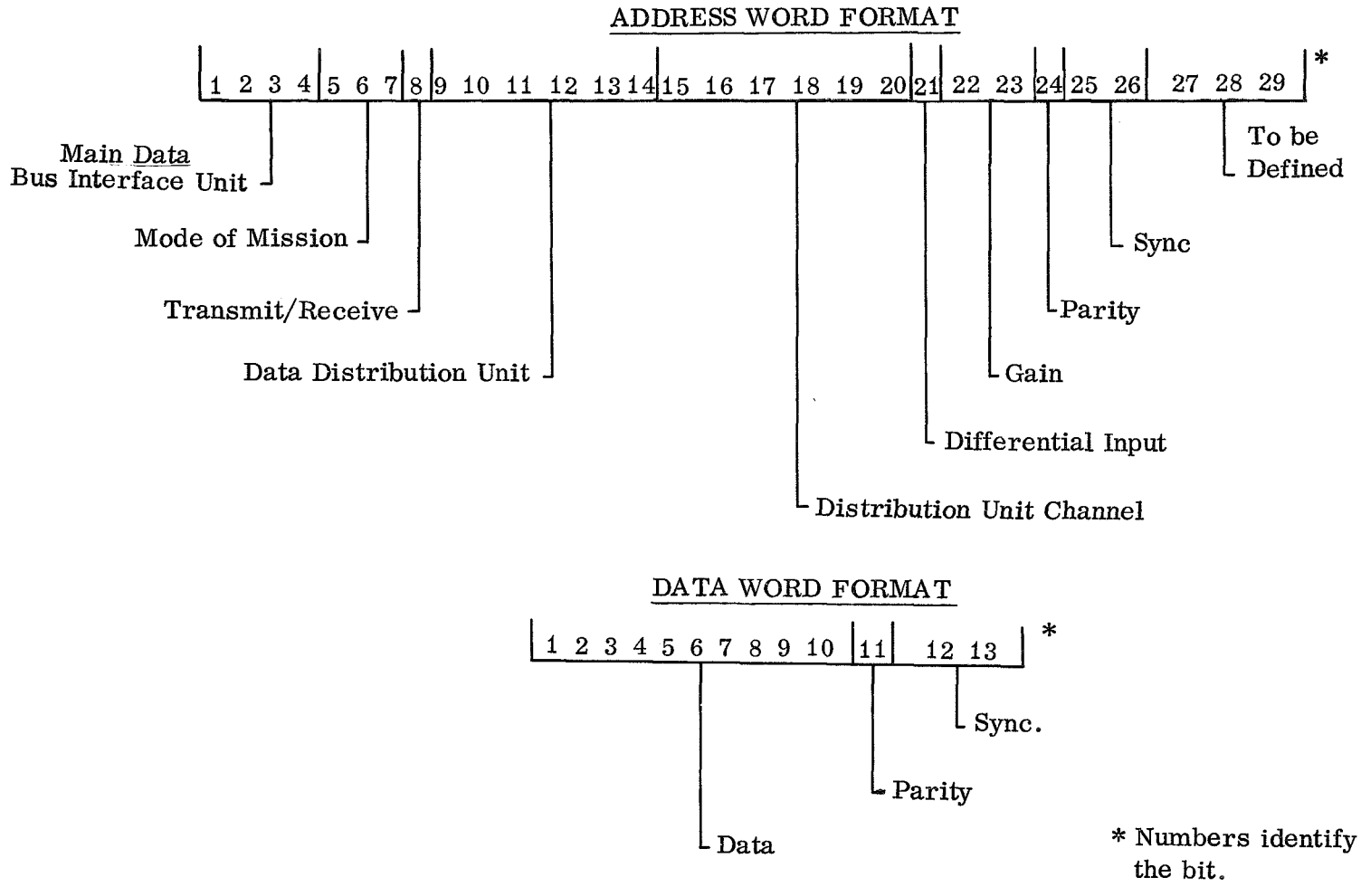


Figure 8-17. Address and data word format.

The address bit rate for the front data processor is 168 Kb/sec and the data rate is 75 Kb/sec. For the center data processor the address rate is 33 Kb/sec with a data rate of 15 Kb/sec. The rear processor required an address rate of 145 Kb/sec with a data rate of 65 Kb/sec. These rates are based on the adjusted sampling rate of 9 samples/sec and processing 10 discretes/word.

8.2.3 Physical Characteristics

A conservative estimate of the size, weight, and power of the major components associated with the DDS is listed in Table 8-4.

TABLE 8-4. PHYSICAL CHARACTERISTICS OF DATA DISTRIBUTION SYSTEM

Component	Quantity	Size (in.)	Weight (lb)	Power (W)
Master Data Processor	1	6 by 4 by 4	4.0	3.0
Data Processor	3	4 by 4 by 4	2.0	2.0
Data Distribution Unit	37	2 by 2 by 2	0.5	0.5

The selected configuration, as shown in Figure 8-8, would weigh approximately 28 pounds and consume 27 watts of power. These characteristics probably will improve by the time the hardware is developed since the integrated circuitry technology is advancing so rapidly.

8.2.4 Onboard Checkout and Fault Isolation

Two basic modes of checkout are applicable to the shuttle. One mode is inflight monitoring with replacement by electronic switching, using onboard permanent storage of system status for quick replacement after returning to ground landing sites. This could include data such as which unit failed and when it failed. The other is a more detailed prelaunch checkout such that hardware may be verified or replaced prior to launch. The onboard tests include fully automatic checking of functions to the level of a black box that can be replaced. The selected configuration allows this by proper programming of the computer, master processor, and the data processors.

Fault isolation is generally included in the automatic checkout; however, more detailed isolation may be performed by the crew through the use of the DDS. The crew has access to the computer which allows any measurement to be displayed or processed by the computer or any other subsystem at any time. The general subject of onboard checkout and evaluation was only lightly considered in this report. A detailed evaluation and analysis will be performed at a later date.

8.3 Computer and Software

8.3.1 Introduction

This section establishes an initial definition and sizing of a computer configuration to support the functional requirements of the subsystems. The problem must be approached from three points as follows:

- The functional requirements of the subsystems.
- The hardware technology for the computer and associated equipment.
- Software and programming.

The three approaches should be pursued in parallel and converge as the design becomes firm (Fig. 8-18). Software and programming are not included in this iteration.

Initial estimates are for the overall computer requirements based on a preliminary definition of subsystem requirements.

8.3.2 Subsystem Requirements

Subsystem requirements are inputs for defining the computer system. The detailed computation and data processing requirements of each subsystem will not be known until later in the design. Therefore, initial estimates were made so that design considerations for this subsystem could proceed along with the other parts of the integrated avionics system. These estimates will be further developed as the other subsystems become better defined. Every subsystem will be involved to some extent. Each subsystem must be evaluated on a continuing basis during the design to establish the definitive subsystem requirements.

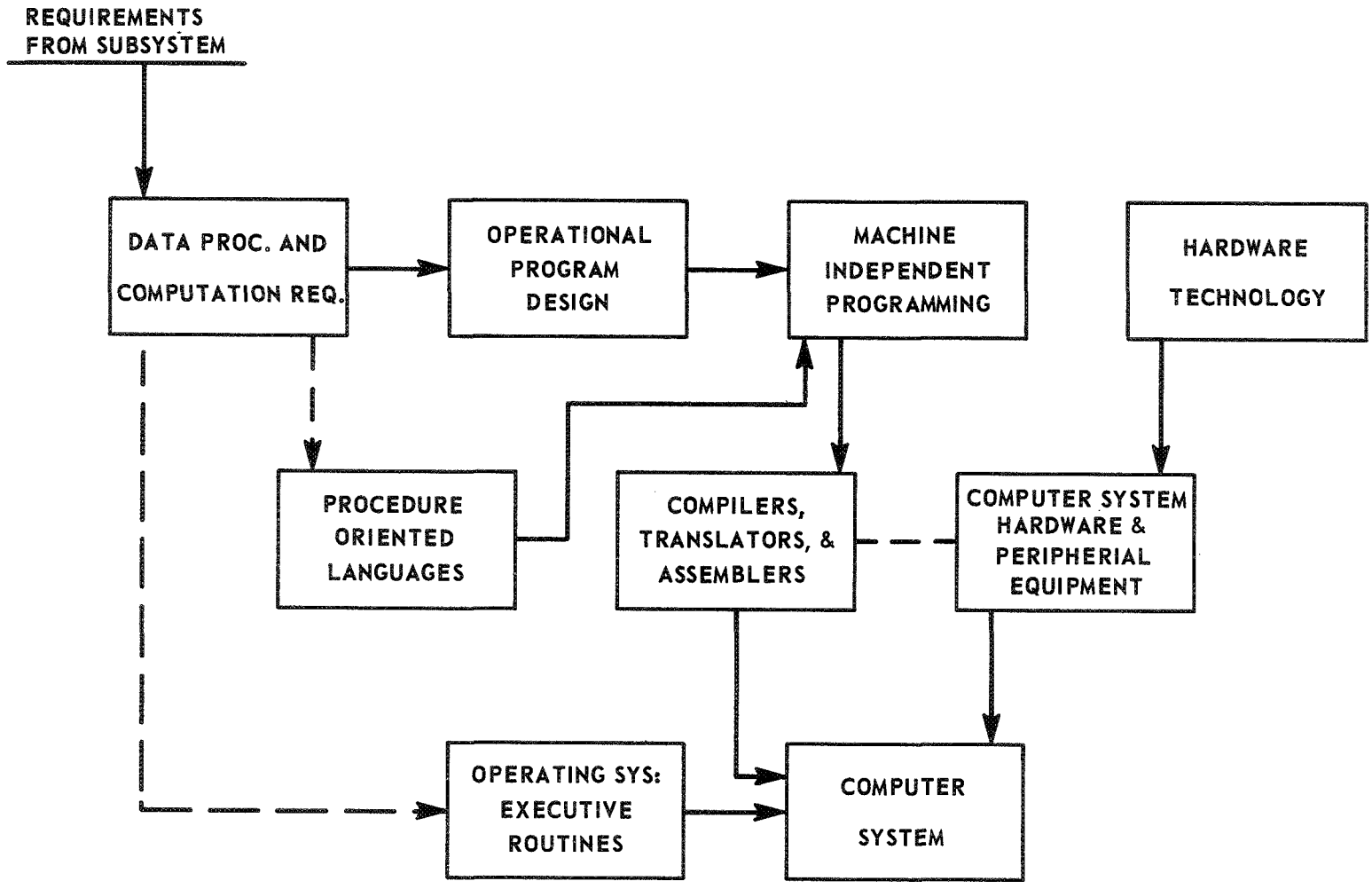


Figure 8-18. Major elements of computer system and software.

To categorize the subsystem requirements for defining and evaluating the functions of the computer and software system, three major function groupings have been identified as follows:

- Checkout, monitoring, and fault isolation.
- Guidance, navigation, and control.
- Operations support.

These functions are required for all mission phases, although the details will be considerably different for different phases. The subsystem performance requirements for each mission phase must be evaluated to establish these functions. The detailed subsystem requirements were evaluated in a qualitative way during this iteration.

8.3.3 Computation and Computer Functions

The major computer functions identified in the previous section, namely, (a) checkout, monitoring, and fault isolation, (b) guidance, control, and navigation, and (c) operations support, are defined in this section.

For the memory estimates in this report, a 32-bit word length is assumed, since this word length is common in computer design and appears to be appropriate for this application.

8.3.3.1 Checkout, Monitoring, and Fault Isolation (CMFI)

These functions are grouped as one, since they are related.

- Checkout refers to the process of evaluating an item of equipment as well as software (computer programs) for operational readiness. Software checkout was not evaluated during this iteration.

- Monitoring refers primarily to a limit checking procedure of all specified test points during operation. It includes any onboard trend analysis required.

- Fault Isolation refers to any diagnostic routines which are sequential procedures for isolating a fault to a single replaceable unit, or to troubleshooting procedures. The requirements for this function are based primarily on the approximately 2000 test points identified in Appendix A.

The program and data storage requirements for CMFI may be categorized under:

- Limit checking.
- Trend analysis.
- Diagnostic routines.

The CMFI computation and data processing storage loads result primarily from the large number of items involved, such as test points, assemblies (black boxes), and replaceable units within each assembly. Fortunately, only a small program is required to process each item. This is in contrast to the guidance, control, and navigation function which requires extensive computations for a few sensor outputs. The number of storage locations for the CMFI may be approximated as follows:

$$\text{Storage locations} = a_1 + a_2 n \quad ,$$

where:

a_1 is the number of basic program logic words associated with the function, a_2 is the average number of data and program words associated with each item,

and

n is the number of items (test points, assemblies, and replaceable units).

Limit Checking

Sample programming has determined that about 100 instruction words are required for a limit checking operation [10]; therefore

$$a_1 = 100 \quad .$$

Assuming that two limits and a nominal error checking value are stored,

$$a_2 = 3$$

Then, for n = 2000 test points

Storage locations: 100 + 3(2000) = 6K

Trend Analysis

Trend analysis will generally require curve fitting. A least squares curve fit program would require approximately 200 instruction words, therefore

$a_1 = 200$.

Assuming 20 data points involved for each n gives

Storage locations = 200 + 20 n .

For trend analysis on every data point, a large memory is needed. Assuming that 10 percent of the measurement points will require trend analysis.

n = 200 .

This means that

Storage locations = 200 + 20(200) = 4K

Diagnostics

The automation of extensive diagnostics in a computer may not be practical for the space shuttle, since every possible situation cannot be predicted. The C-5 aircraft MADAR monitoring system made all onboard diagnostics manual. The space shuttle is estimated to have approximately 80 assemblies. (black boxes). Each assembly will have a specified number of replaceable units. The number of replaceable units (RU) has not been determined, but 5 to 10 per assembly appears reasonable; however, preliminary evaluation by sample programming indicates that 500 basic program instructions are required for a diagnostic routine. Assuming n is the number of replaceable units per assembly, 500 is the number of basic program instructions, and 10 is the number of data points per RU, the memory requirements are determined as follows:

Memory locations = 500 + 10 n per assembly.

Then, for 80 assemblies the total memory locations are estimated by:

$$\text{Memory locations} = 80(500 + 10n).$$

If n ranges between 5 and 10 replaceable units per assembly, the memory requirements for 80 assemblies will be between 44 and 48K. Due to indicated trends in onboard checkout, 8 assemblies (10 percent of total) was selected as typical for this task. So, the estimate for this task is for $n = 5$.

$$\text{Memory locations} = 8(500 + 10 \times 5) = 4.4\text{K}.$$

The executive program for checkout will require about 10 percent of the other memory locations. For CMFI, the total memory requirements are

Limit checking	6K
Trend analysis	4K
Diagnostics	4.4K
Subtotal	14.4K
Executive Program	1.4K
	15.8K

This figure is rounded off to 16K.

8.3.3.2 Guidance, Navigation, and Control (GN&C)

Techniques for utilizing digital computations and computers for this subsystem are well known for both space vehicles and avionics. The preliminary analyses that were performed lead to the conclusion that an extensive computational capability will be required for this subsystem. Section 5.0 indicated that the GN&C subsystem will utilize an inertial measurement unit (IMU) during all phases of the mission and may require a solution of the equations of motion on board. Except for ascent and reentry, all other mission phases require 3, 4, or 5 navigation instruments to support the IMU. Use of landmark and star trackers for navigation requires sizable computer storage for storing ephemeris and landmark location data as well as considerable processing and computation.

The type of guidance and navigation equation to be used is uncertain for all phases except ascent. For ascent, some type of optimal guidance and navigation will probably be used. For example, Saturn V ascent requires about 12K simplex memory locations for the basic flight program which includes navigation, guidance, control, telemetry, and some sequencing.

Digital filtering will place a further requirement on the computer system. For example, digital control for Titan III utilizes a 4K memory. These considerations plus allocations for rendezvous and other mission requirements lead to a baseline memory allocation of 32K for the stored program and data of the GN&C subsystem.

If a strapdown IMU is used, a small single processor with about 2 to 4K memory will be dedicated to the IMU. This allocation is in addition to the 32K above.

8.3.3.3 Operations Support

Operations support embraces any computer tasks that are required to assist the flight crew in the real-time operation of the vehicle during all mission phases, as well as ground support between missions.

The majority of the tasks may be defined as belonging to the following:

- Abort warning.
- Mission planning.
- Subsystem control.

These are defined below:

Abort warning provides the crew with monitoring of mission critical parameters by evaluation of basic data from the CMFI function. Typically, 200 test points will be utilized. This requires a program storage of about 1K memory locations.

Mission planning provides the crew with a capability to alter missions and schedules on board the vehicle. This task will probably not be used frequently. The programs may be stored in bulk storage until needed; however, considerable trajectory calculations are necessary and can easily require most of a 16K memory during use.

Subsystem control and sequencing is a combination of computer support for all subsystems except GN&C. Considerable subsystem definition is required as inputs for this task.

The following are general requirements to be expected from the subsystems.

- Prepare to Launch Routines (Saturn V uses about 4K memory location for this).

- Data distribution system control.

- Sequencing events, such as shutting down and starting engines during staging, and initiating other subsystem start-up and shutdown.

- All computation and processing required for control and operation of other subsystems, such as environmental control, electrical power, propellant management, antenna pointing, and updating displays.

To perform subsystem control and sequencing, data associated with approximately 500 measurement and control points (25 percent of measurement points in Appendix A) will be utilized.

Considering the above tasks over all mission phases and noting that all tasks will not be performed concurrently, it is estimated that a main memory requirement of 16K is adequate.

8.3.4 Computer System Configuration For Space Shuttle

8.3.4.1 Introduction

The major elements of a computer — processor, memory, and input-output may be organized in various ways to meet requirements of a vehicle system, such as a space shuttle. Preliminary evaluation of computer hardware technology for 1972 indicates that computers will meet the speed requirements of a space shuttle vehicle system. The same technology indicates that the weighting of design criteria for the space shuttle computer system will probably be different from the weighting used in the past. Satisfactory performance can be achieved for reasonable power, weight, and volume allocations. Reliability and cost design criteria will become predominant. For a given performance some power, weight, and volume can be traded for reliability and probably cost.

Maintenance is mandatory for a long-term operational vehicle such as a space shuttle. Maintenance is intricately related to reliability and cost. For a long-term operating system, total cost must be considered a function of at least two subdivisions: — initial cost (including development cost) and operating cost. Since cost is difficult to establish in preliminary evaluations, reliability will predominate in the early trade studies. Maintenance concepts will be considered, although maintenance policies and procedures are not established at this time.

Four types of computer organizations have been selected as basic candidates for the computer system configuration of the space shuttle. In the following sections the four types are defined and discussed. An initial configuration for the shuttle computer system is selected based on the estimated requirements from previous sections and tentative reliability considerations.

8.3.4.2 Configurations of Computer Systems

The configurations described in this section were selected as typical for a starting point; one was finally selected as a baseline. As the performance and reliability evaluations progress, other combinations may evolve or become apparent.

Some characteristics of each type of configuration are stated and discussed. While this discussion and these characteristics are not exhaustive, they represent an introduction to the system considerations involved. Trade-offs for each configuration should continually be made, using the same set of subsystem requirements.

Uniprocessor — A single processor unit (PU) which may utilize one or more memory modules. This is a basic computer configuration. The processor unit may have extensive capability and several memory modules or it may be small and dedicated to a particular job (Fig. 8-19).

Federated Computers — Several uniprocessors are assigned and dedicated to one or more subsystem functions. Required memory units are controlled by each uniprocessor (Fig. 8-20). The data bus is the only path for data transmission among the processor units. In general, for a particular application one of the uniprocessors may be replaced by more than one processor.

Dual Processors — Two processor units and their associated memory units are permitted intercommunication between processors. Two levels of intercommunications are illustrated in Figures 8-21A and 8-21B.

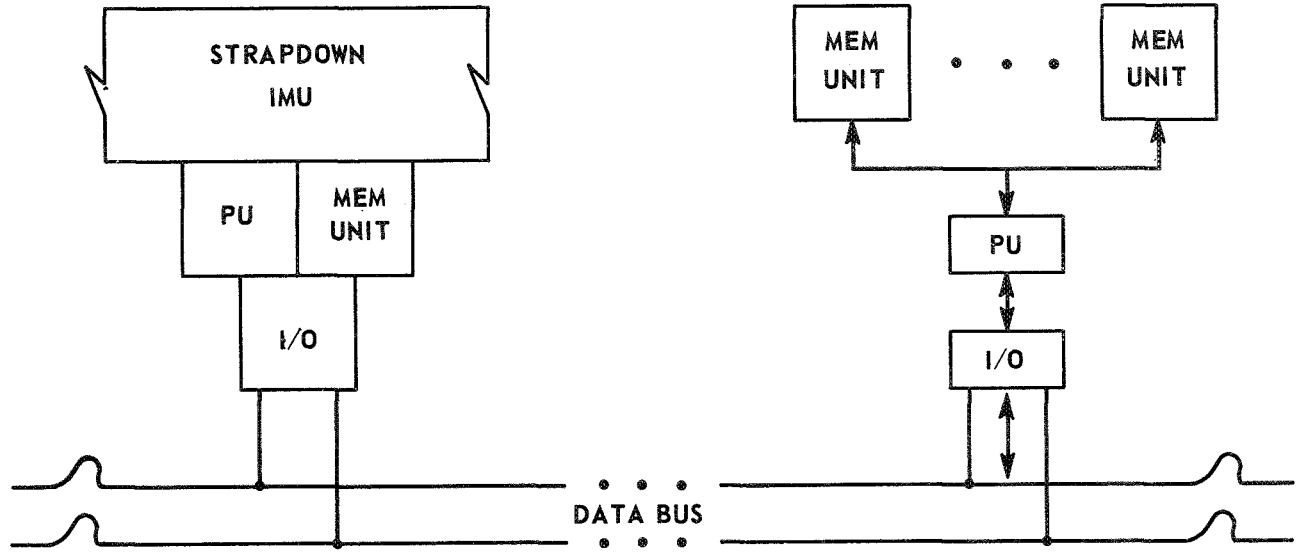
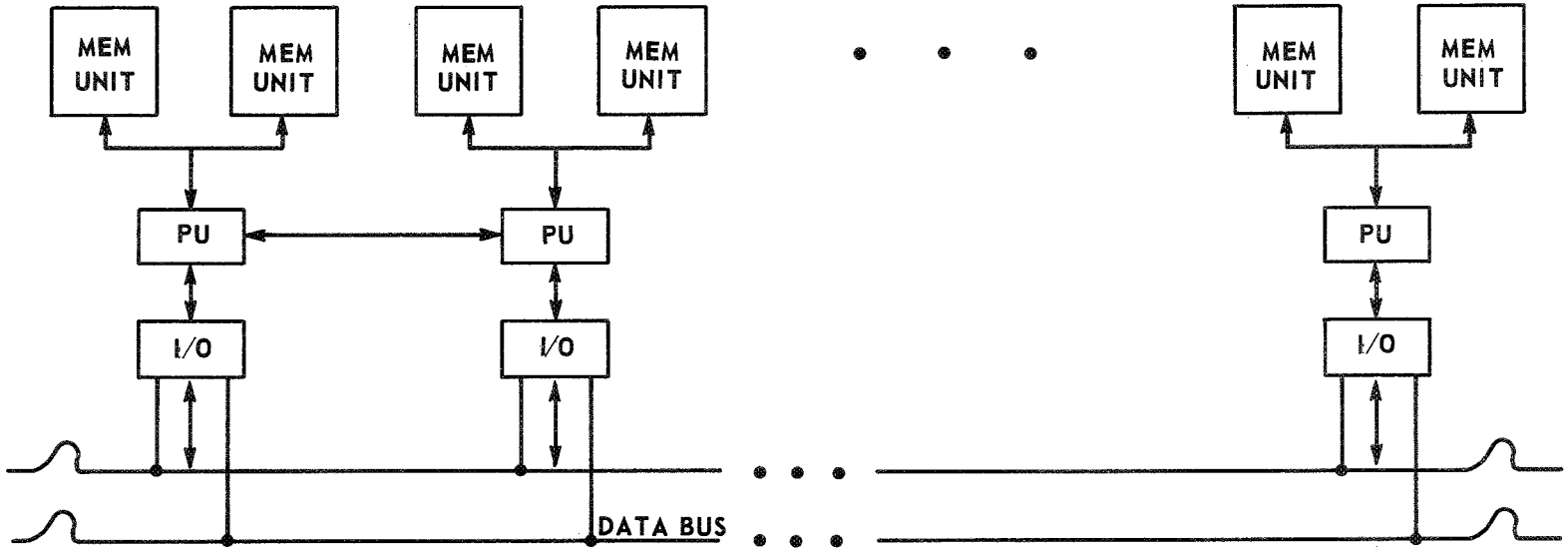


Figure 8-19. Uniprocessor configuration.



- SEPARATE GENERAL PURPOSE COMPUTERS
- INTER-COMPUTER DATA MULTIPLEXED ON COMMON DATA BUS
- COMPUTERS HAVE COMMON PU AND MEMORIES
- SAME REDUNDANCY REQUIRED IN EACH COMPUTER
- DEGRADED MODE CAPABILITY IS POOR
- REQUIRES SEPARATE PROGRAM FOR EACH COMPUTER
- MANY PROGRAMS OR ROUTINES WOULD BE DUPLICATED IN EACH COMPUTER

Figure 8-20. Federated computer.

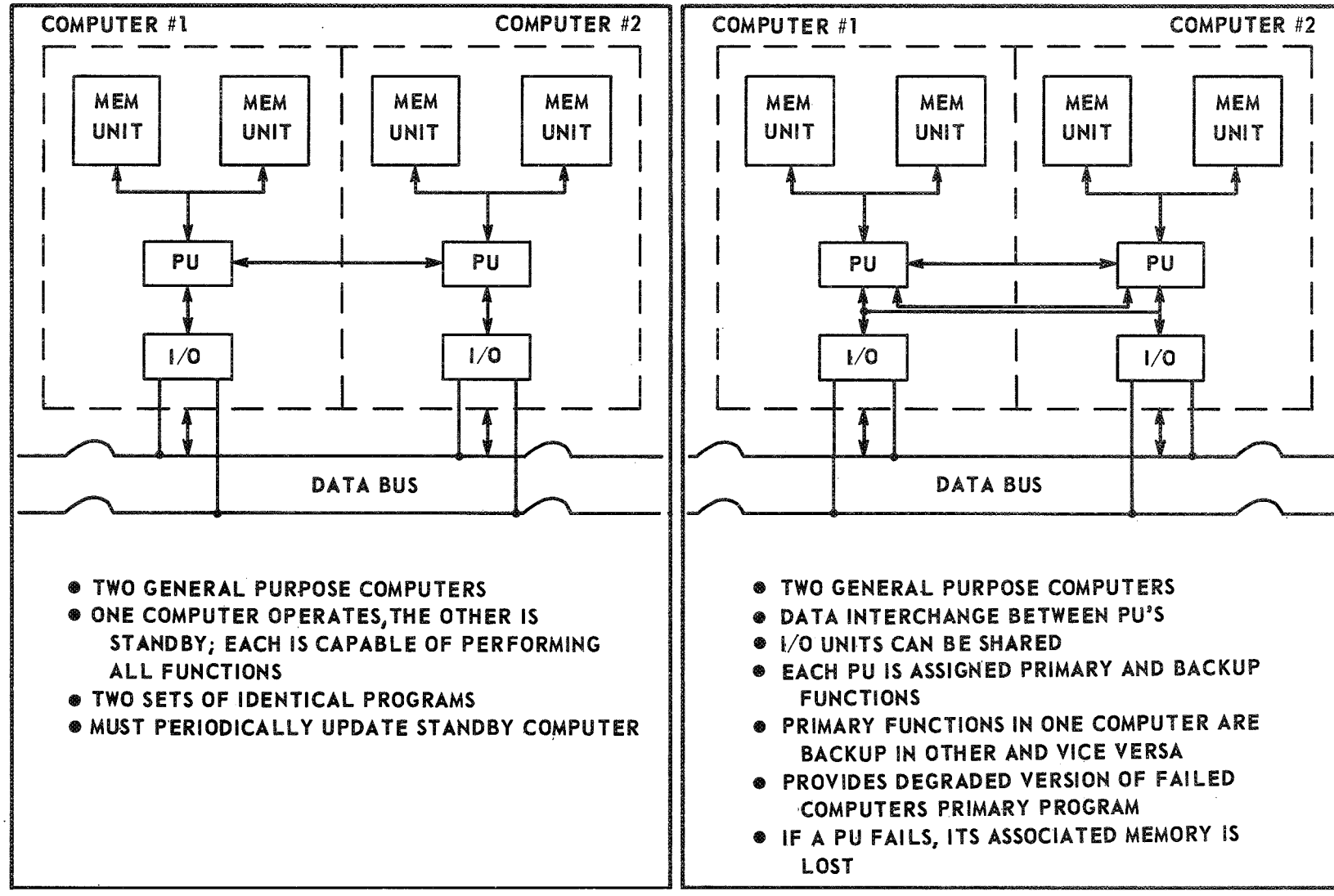


Figure 8-21. Dual processors configuration.

Multiprocessor — Two or more processor units have access to any memory unit or input/output unit. Although a multiprocessor is not rigidly defined it refers, in general, to a multiple processor configuration (Fig. 8-22) in which any processor may be assigned any program residing in any memory unit. The assignment is made by an executive program depending on which processor is idle.

8.3.4.3 Selected Computer System for the Space Shuttle

Computer hardware technology predictions for 1972 indicate that basic computer characteristics listed below are feasible. These typical processor characteristics are for a simplex machine and include the power supply:

- Modular replaceable units.
- A parallel machine with floating point number presentation .
- Approximately 75 instructions.
- Each processor capable of memory expansion to 64K.
- 32-bit word length capability.
- Add time — 1-2μsec
- Multiply time — 8μsec
- Divide time — 10μsec
- Memory cycle time — 0.5μsec
- Volume with 8K memory — 1 cubic foot.
- Weight — 5 pounds
- Approximate power — 15 watts.

The following guidelines were assumed for the conceptual design.

— A central location of the main computer is practical, since total data rates are about 500Kb/sec and this is within the capacity of the data bus.

— All limit checking for test point monitoring will be performed in the main computer (see Section 8.3.3.1). Limit checking may later prove to be more practical at or near the source of the data.

— Small computers, independent of the main computer, may be used because of specialized computation requirements at specific locations illustrated in Figure 8-19.

— Software and operating characteristics will influence the system selection.

The common denominator for selecting the computer configuration was reliability.

The uniprocessor (Fig. 8-19) with appropriate memory units was not selected due to the uncertainty of meeting required reliability. The federated computer (Fig. 8-20) would require more weight and power to provide the redundancy necessary for reliability. The dual processor configurations (Fig. 8-21) utilize the standby concept, and could probably be used successfully; however, their reliability advantage over the shared memory configuration is not decisive.

A centrally located multiple processor (CPU) configuration with a small dedicated computer (Fig. 8-23) was selected for this conceptual design. In this configuration each processor unit is assigned a task and has its own executive program; i.e., one processor unit is assigned the operations support function, one processor unit the checkout, monitoring, and fault isolation, and the other processor unit the guidance, control, and navigation. Degraded mode configuration and operation will be provided, principally by software. Preliminary results of other investigators [2, Vol V] indicate that this type of configuration represents the best approach within reliability, weight, power, and other trade areas.

Each PU operates only on the tasks assigned to it and under control of its own executive program. Failure of any one unit (PU, memory, or input/output) will not cause the computer system to be incapacitated. An overall failure mode executive program will decide which functions are to be performed and at what level, if a PU or memory unit fails. The total of these executive programs will be less complex than a single overall executive program. This is due to the delineation of functions among the four executive programs. This separation of functions will also lower the probability of two PU's trying to access the same memory location.

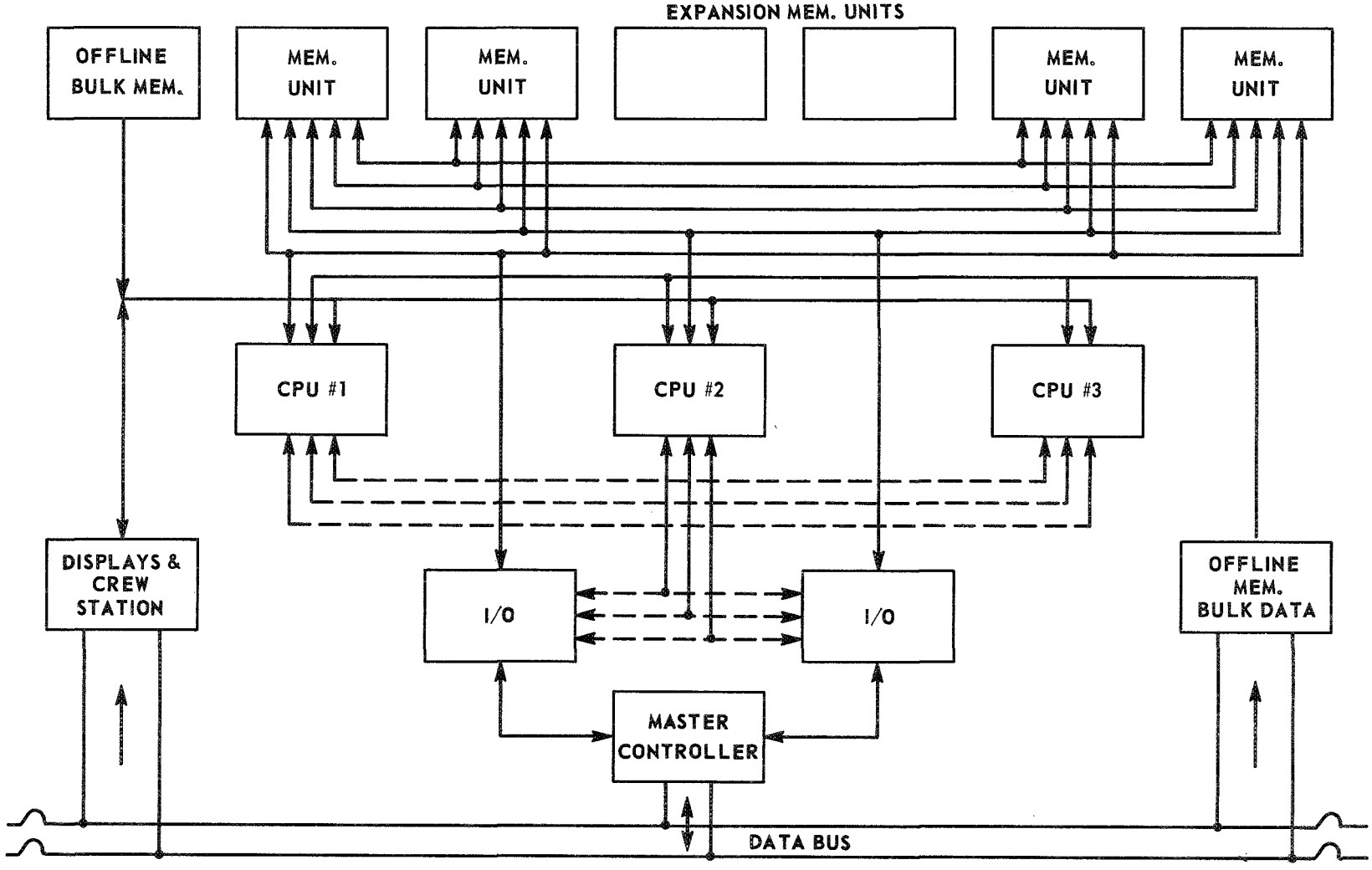


Figure 8-22. Multiprocessor configuration.

In summary this selected configuration has the following two advantages over the classical multiprocessor configuration, in which any processor unit shares any job:

- Software is less complex, since one executive program does not control the entire computer operation.

- There is lower probability of more than one processor unit accessing the same memory concurrently.

8.3.5 Summary of Space Shuttle Computer System

In addition to the memory sizing for data and stored programs the duty cycle of the subsystem places a speed requirement on the processor units. Preliminary analyses to date indicate that 1972 technology will be adequate.

The memory capacity for the computer system is summarized in Table 8-5. In this table the 32K for contingencies and expansion is broken down as follows:

- 8K represents 10 to 15 percent of main memory for executive programs.

- 8K represents compiler inefficiencies due to higher level languages.

- 16K represents reserve for expansion.

TABLE 8-5. COMPUTER SYSTEM MEMORY CAPACITY

Checkout, Monitoring, and Fault Isolation	16K × 32 bits
Guidance, Navigation, and Control	32K × 32 bits
Operations Support	16K × 32 bits
Contingencies and Expansion	32K × 32 bits
Total	96K × 32 bits

A bulk memory, such as a magnetic tape unit, will be required as indicated in Figure 8-23, although its requirements were not evaluated in detail. A capacity of 20 M bits was estimated to be adequate for this initial iteration. The computer dedicated to the strapdown IMU is allocated a 4K by 32-bit memory unit. The physical characteristics are summarized in Table 8-6.

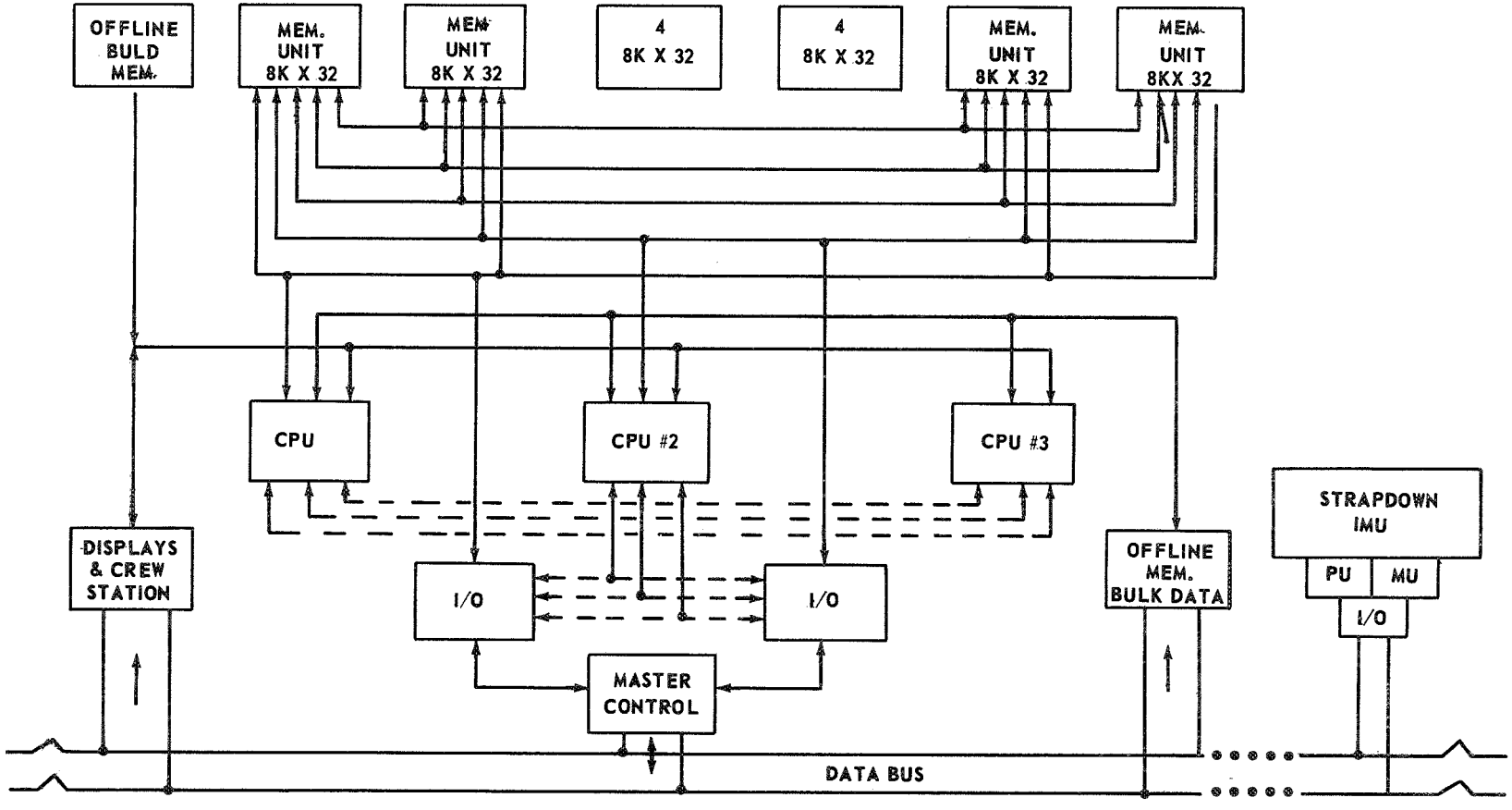


Figure 8-23. Conceptual design computer system configuration.

8.3.6 Recommendations

The following recommendations are made:

- A software baseline be established by evaluation of both onboard and ground support requirements. The ground support computer baseline should be included.

- A simulation be initiated to evaluate the tradeoffs and integration of computers, software, crew, and vehicle subsystems.

TABLE 8-6. PHYSICAL CHARACTERISTICS OF CONCEPTUAL DESIGN COMPUTER SYSTEM

Computer	Quantity	Weight (lb)	Power (W)	Volume (cu ft)
8K MU	12	30	120	0.6
PU	3	10	60	0.3
I/O	2	5	20	0.1
Bulk Mem. Unit	1	5	20	0.1
Subtotal		50	220	1.1
Strapdown Computer	1	5	20	0.1
Total		55	240	1.2

APPENDIX A

SHUTTLE SYSTEM SIGNAL CHARACTERISTICS

The data requirement list compiled in this appendix, with the exception of Section 3.0, was extracted from "IES Data Requirements" by Lockheed Missiles and Space Company, Sunnyvale, California, dated October 17, 1969. Electrical Power Subsystem data requirements (Section 3.0) were compiled by PD-DO-EP.

The data points are grouped according to subsystem. A functional description of each data point is listed with detailed characteristics of each data point. A tabulation is provided showing data origin, termination, and test point requirement.

1.0 Structural/Mechanical

Data Points	Input From	Output To	Test Point	Characteristics					
				Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
1.1 HYDRAULIC SUPPLY & CONTROL									
Pump Motor Cont.	8.0		X	On/Off	dc		24V	—	
Pump Motor Press.		7.0, 8.0	X	Anal	100 Hz		0-3000 psi	±25 psi 0-5V	
Pump Motor Temp.		7.0, 8.0	X	Anal	10 Hz		0 to +200° F	±10° F 0-5V	
APU Cont.	7.0	7.0	X	On/Off	dc		24V	—	
APU Turbine Output Temp.		7.0, 8.0	X	Anal	5 Hz		0-1500° F	±25° F 0-5V	
APU Turbine Output Press.		7.0, 8.0	X	Anal	100 Hz		0-3000 psi	±25 psi 0-5V	
GO_2 Press.		7.0, 8.0	X	Anal	10 Hz		20 to 700 psi	±10 psi 0-5V	
GO_2 Temp.		8.0	X	Anal	5 Hz		-296° to 100° F	±10° F 0-5V	
GH_2 Press.		7.0, 8.0	X	Anal	10 Hz		20 to 700 psi	±10 psi 0-5V	
GH_2 Temp.		8.0	X	Anal	5 Hz		-423° to 100° F	±10° F 0-5V	
APU Combustion Press.		7.0, 8.0	X	Anal	50 Hz		0 to 500 psi	±10 psi 0-5V	
APU Combustion Temp.		8.0	X	Anal	20 Hz		0 to 2000° F	±25° F 0-5V	
APU Exhaust Temp.		7.0, 8.0	X	Anal	10 Hz		0 to 1000° F	±25° F 0-5V	
APU Exhaust Press.		7.0, 8.0	X	Anal	5 Hz		0 to 30 psi	±2 psi 0-5V	
Gas Turbine Speed		7.0, 8.0	X	Dig.	0-10 KPPS		0-50 Krpm	±100 rpm 5V	
Gas Turbine Change in Speed		8.0	X	Dig.	0-2 KPPS		0-±2K rpm/sec	±10 rpm per sec 5V	
Gearbox Temp.		8.0	X	Anal			0-500° F	±10° F 0-5V	
2-Way Latching Valve Cont		7.0		On/Off	dc		Pos. 1 or 2	— 24V	
Latching Valve Pos.		8.0	X	On/Off	dc		Pos. 1 or 2	— 5V	
Accumulator Precharge		7.0		On/Off	dc		On or Off	— 24V	
Accumulator Temp.		8.0	X	Anal	10 Hz		0-200° F	±10° F 0-5V	
Accumulator Press.		7.0, 8.0	X	Anal	50 Hz		0-3000 psi	±25 psi 0-5V	

Data Points	Input From	Output To	Test Point	Characteristics					
				Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
1.2 PRIMARY ENGINE GIMBAL (PER GIMBAL)									
Gimbal Pos		5.0	X	Anal	400	3	±8°	0.1%	
Gimbal Rate		5.0	X	Anal	400	3	±3°/sec	0.1%	
Gimbal Cmd	5.0		X	Anal		>30	±10	0.1%	
Differential Press.		8.0	X	Anal		3	±5Kpsi		
Temp.		8.0	X	Anal		0.1			
Flow		8.0	X	Dig.		0-1K PPS	1GPM		
Voltage		8.0	X	Anal		0.1	30V		
Current		8.0	X	Anal	30	0.1	±10ma		
1.3 VEHICLE SEPARATION CONTROL									
Vehicle Release Arming	5.0, 8.0		X	On/Off	dc	dc	24V	—	3 each
Vehicle Release Command	5.0, 8.0			On/Off	dc	dc	24V	—	3 each
Firing Unit Temp.		8.0	X	Anal	2 Hz	1 Hz	0-250° F	±5° F	3 each
Firing Unit Test	8.0			On/Off	dc	dc	24V	—	3 each
Firing Unit Output		8.0	X	Pulse	2 kc	500 Hz	0-2000V	—	3 each
Squib Temp.		8.0	X	Anal	1 Hz	1 Hz	0-500° F	±10° F	3 each
Squib Detonate Sensor		8.0	X	Pulse	1 kc	100 Hz	0-3000° F	—	3 each
Separation (Continuity)		5.0, 8.0	X	On/Off	dc	dc	0 or Inf. ohms	—	3 each
Flyaway Rocket Arming	5.0, 8.0			On/Off	dc	dc	24V	—	3 each
Fire Flyaway Rocket		8.0	X	On/Off	dc	dc	24V	—	3 each
Firing Unit Temp		8.0	X	Anal	2 Hz	1 Hz	0-250° F	±5° F	3 each
Firing Unit Test	8.0			On/Off	dc	dc	24V	—	3 each
Firing Unit Output		8.0	X	Pulse	2 kc	500 Hz	0-2000V	—	3 each
Squib Temp.		8.0	X	Anal	1 Hz	1 Hz	0-500° F	±10° F	3 each
Squib Detonate Sensor		8.0	X	Pulse	1 kc	100 Hz	0-3000° F	—	3 each
Separation (Continuity)		5.0, 8.0	X	On/Off	dc	dc	0 or Inf. ohms	—	3 each
1.4 NOSE CAP CONTROL									
Nose Pos. Command Interlock	5.0, 8.0			On/Off	dc	dc	24V	—	

Data Points	Input From	Output To	Test Point	Characteristics					
				Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
Nose Cap Translation Command	5.0, 8.0			On/Off	dc	dc	24V	—	
Cap Actuation Controller Output		8.0	X	Anal	400 Hz	5 Hz	0-115V	±2V	
Cap Act. Controller Temp.		8.0	X	Anal	1 Hz	1 Hz	0-250°F	±10°F 0-5V	
Nose Pos. Sensor		7.0, 8.0	X	Anal	5 Hz	5 Hz	0-18 in.	±0.2 in. 0-5V	
Nose Closure Switch		7.0	X	On/Off	dc	dc	5V	—	
Cap Actuation Motor Temp.		8.0	X	Anal	2 Hz	2 Hz	0-500°F	±10°F 0-5V	
Heat Shield Closure		7.0	X	On/Off	dc	dc	5V	—	
Gear Train Temp.		8.0	X	Anal	5 Hz	5 Hz	0-500°F	±20°F 0-5V	
Gear Train Pos.		8.0	X	Anal	5 Hz	5 Hz	0-X00 Rev's	±X Rev's 0-5V	
1.5 THRUSTER DEPLOYMENT									
Thruster Pos. Command Interlock	5.0, 8.0			On/Off	dc	dc	24V	—	
Deployment/Retraction Initiation Command	5.0, 8.0			On/Off	dc	dc	24V	—	
Actuator Controller Output		8.0	X	Anal	400 Hz	5 Hz	0-115V	±2V	
Actuator Controller Temp.		8.0	X	Anal	1 Hz	1 Hz	0-250°F	±10°F 0-5V	
Thruster Pos.		7.0, 8.0	X	Anal	5 Hz	1 Hz	0-100°F	±2° 0-5V	
Thruster Lock/Limit		7.0	X	On/Off	dc	dc	5V	—	
Deployment Motor Temp.		8.0	X	Anal	1 Hz	1 Hz	-60-200°F	±10°F 0-5V	
Deployment Motor Speed		8.0	X	Dig.	0-10 KPPS	0-1 KPPS	0-200 rpm	±20 rpm 0-5V	
Gear Train Temp.		8.0	X	Anal	1 Hz	1 Hz	-60-+300°F	±20°F 0-5V	
Gear Train Pos.		8.0	X	Anal	20 Hz	5 Hz	0-20 in.	±1/2 in. 0-5V	
Door Pos.		7.0, 8.0	X	Anal	20 Hz	5 Hz	0-100°F	±2° 0-5V	
Door Lock/Limit		7.0	X	On/Off	dc	dc	5V	—	
Gear Train Temp.		8.0	X	Anal	1 Hz	1 Hz	-60-+300°F	±20°F 0-5V	
Gear Train Pos.		8.0	X	Anal	20 Hz	5 Hz	0-12 in.	±1/2 in. 0-5V	
Heat Shield Closure (Seal)		7.0, 8.0	X	On/Off	dc	dc	5V	—	

Data Points	Input From	Output To	Test Point	Characteristics					
				Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
1.6 DOCKING ADAPTER CONTROL									
Docking Initiation (Arm)	5.0, 6.0			On/Off	dc	dc	24V	—	
Probe Deployment Command	5.0, 6.0			On/Off	dc	dc	24V	—	
Probe Actuation Command	5.0, 6.0			On/Off	dc	dc	24V	—	
Probe Orientation		8.0	X	Anal	2 Hz	2 Hz	0-36 in.	±1 in. 0-5V	
Probe Length		8.0, 7.0	X	Anal	2 Hz	2 Hz	0-50 ft	±1/2 ft 0-5V	
Probe — Drogue Contact		8.0, 7.0	X	On/Off	dc	dc	0 ohm	20 ohms	
Probe Depl. Actuator Temp.		8.0	X	Anal	1 Hz	1 Hz	0-500°F	±20°F 0-5V	
Probe Depl. Actuator Press.		8.0	X	Anal	5 Hz	5 Hz	0-3000 psi	±25 psi 0-5V	
Probe Extension Actuator Temp.		8.0	X	Anal	1 Hz	1 Hz	0-500°F	±20°F 0-5V	
Probe Extension Actuator Press.		8.0	X	Anal	5 Hz	5 Hz	0-3000 psi	±25 psi 0-5V	
Probe Temp		8.0	X	Anal	4 Hz	4 Hz	0-500°F	±20°F 0-5V	
Docking Cont. Electronics Temp.		8.0	X	Anal	1 Hz	1 Hz	-60-200°F	±10°F 0-5V	
Docking Cont. Electronics Output 1		8.0	X	Anal	2 Hz	2 Hz	0-36 in.	±1 in. 0-28V	
Docking Cont. Electronics Output 2		8.0	X	Anal	2 Hz	2 Hz	0-50 ft	±1/2 ft 0-28V	
Docking Cont. Electronics Output 3		8.0	X	Anal	2 Hz	2 Hz	0-10 ft	±3 in. 0-5V	
Docking Cont. Electronics Output 4		8.0	X	Anal	2 Hz	2 Hz	0-110°	±5° 0-28V	
Docking Cont. Electronics Output 5		8.0	X	Analog	2 Hz	2 Hz	0-20 ft 0-70 ft	±1/2 ft 0-28V	
Cargo Doors Actuate Cmd.	5.0, 6.0 7.0, 8.0			On/Off	dc	dc	24V	—	
Cargo Doors Actuator Temp.		8.0	X	Anal	1 Hz	1 Hz	0-300°F	±20°F 0-5V	
Cargo Doors Actuator Press.		8.0	X	Anal	1 Hz	1 Hz	0-3000 psi	±25 psi 0-5V	
Cargo Doors Pos.		8.0, 7.0	X	Anal	2 Hz	2 Hz	0-110°	±5° 0-5V	
Cargo Docking Actuate Cmd	5.0, 6.0 7.0, 8.0			On/Off	dc	dc	24V	—	

Data Points	Input From	Output To	Test Point	Characteristics					
				Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
Cargo Docking Actuators Temp.		8.0	X	Anal	1 Hz	1 Hz	0-300°F	±10°F 0-5V	
Cargo Docking Actuators Press.		8.0	X	Anal	1 Hz	1 Hz	0-3000 psi	±25 psi 0-5V	
Cargo Pos.		8.0,7.0	X	Anal	2 Hz	2 Hz	0-20 ft	±1/2 ft 0-5V	
Cargo Lock State		8.0,7.0	X	On/Off	dc	dc	0 ohm	20 ohms	
Cargo Transfer Pos.		8.0,7.0	X	Anal	2 Hz	2 Hz	0-70 ft	±1/2 ft 0-5V	
1.7 WING DEPLOYMENT									
Landing Phase Initiation (Arm)	5.0,6.0 7.0,8.0			On/Off	dc	dc	24V	—	
Open Wing Doors Command	5.0,6.0 7.0,8.0			On/Off	dc	dc	24V	—	
Wing Deployment Controller Temp.		8.0	X	Anal	1 Hz	1 Hz	0-200°F	±5°F 0-5V	
Wing Deployment Controller Output 1		8.0	X	Anal	2 Hz	1 Hz	0-90°	±2° 0-5V	
Wing Deployment Controller Output 2		8.0	X	Anal	2 Hz	1 Hz	0-110°	±5° 0-5V	
Wing Deployment Actuator Temp.		8.0	X	Anal	4 Hz	2 Hz	0-1500°F	±20°F 0-5V	
Wing Deployment Actuator Press.		8.0	X	Anal	4 Hz	2 Hz	0-3000 psi	±25 psi 0-5V	
Right Door Actuator Temp.		8.0	X	Anal	4 Hz	2 Hz	0-1500°F	±20°F 0-5V	
Right Door Actuator Press.		8.0	X	Anal	2 Hz	1 Hz	0-3000 psi	±25 psi 0-5V	
Left Door Actuator Temp.		8.0	X	Anal	4 Hz	2 Hz	0-1500°F	±20°F 0-5V	
Left Wing Temp (Leading)		8.0	X	Anal	10 Hz	5 Hz	0-2500°F	±20°F 0-5V	
Left Wing Temp. (Tip)		8.0	X	Anal	10 Hz	5 Hz	0-2500°F	±20°F 0-5V	
Left Wing Temp. (Bottom)		8.0	X	Anal	10 Hz	5 Hz	0-2500°F	±20°F 0-5V	

Data Points	Input From	Output To	Test Point	Characteristics					
				Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
Right Wing Temp (Leading)		8.0	X	Anal	10 Hz	5 Hz	0-2500°F	±20°F 0-5V	
Right Wing Temp (Tip)		8.0	X	Anal	10 Hz	5 Hz	0-2500°F	±20°F 0-5V	
Right Wing Temp (Bottom)		8.0	X	Anal	10 Hz	5 Hz	0-2500°F	±20°F 0-5V	
Left Wing Door Pos.		7.0, 8.0	X	Anal	2 Hz	1 Hz	0-110°	±5° 0-5V	
Left Wing Pos.		5.0 7.0, 8.0	X	Anal	2 Hz	1 Hz	0-90°	±2° 0-5V	
Right Wing Door Pos.		7.0, 8.0	X	Anal	2 Hz	1 Hz	0-110°	±5° 0-5V	
Right Wing Pos.		5.0 7.0, 8.0	X	Anal	2 Hz	1 Hz	0-90°	±2° 0-5V	
Left Door Actuator Press.		8.0	X	Anal	2 Hz	1 Hz	0-3000 psi	±25 psi 0-5V	
1.8 LANDING GEAR CONTROL									
Open Nose Doors Command	5.0 7.0, 8.0			On-Off	dc	dc	24V	—	
Landing Gear Down Command	5.0 7.0, 8.0			On/Off	dc	dc	24V	—	
Nose Gear Controller Temp.		8.0	X	Anal	1 Hz	1 Hz	0-200°F	±5°F	
Nose Gear Controller Output 1		8.0	X	Anal	4 Hz	2 Hz	0-90°	±2°	
Nose Gear Controller Output 2		8.0	X	Anal	4 Hz	2 Hz	0-110°	±5°	
Left Main Gear Controller Temp.		8.0	X	Anal	1 Hz	1 Hz	0-200°F	±5°F	
Left Main Gear Controller Output		8.0	X	Anal	4 Hz	2 Hz	0-90°	±2°	
Right Main Gear Controller Temp.		8.0	X	Anal	1 Hz	1 Hz	0-200°F	±5°F	
Right Main Gear Controller Output		8.0	X	Anal	4 Hz	2 Hz	0-90°	±2°	
Nose Gear Actuator Temp.		8.0	X	Anal	5 Hz	1 Hz	0-1000°F	±20°F	
Nose Gear Actuator Press.		8.0	X	Anal	10 Hz	2 Hz	0-3000 psi	±25 psi	
Nose Door Actuator Temp.		8.0	X	Anal	5 Hz	1 Hz	0-1000°F	±20°F	
Nose Door Actuator Press.		8.0	X	Anal	10 Hz	2 Hz	0-3000 psi	±25 psi	
Left Main Gear Act. Temp		8.0	X	Anal	5 Hz	1 Hz	0-1000°F	±20°F	
Left Main Gear Act. Press.		8.0	X	Anal	10 Hz	2 Hz	0-3000 psi	±25 psi	
Right Main Gear Act. Temp.		8.0	X	Anal	5 Hz	1 Hz	0-1000°F	±20°F	
Right Main Gear Act. Press.		8.0	X	Anal	10 Hz	2 Hz	0-3000 psi	±25 psi	
Nose Gear Wheel Speed		7.0, 8.0	X	Dig.	0-10 KPPS	0-1 KPPS	0-3000 rpm	±20 rpm	

Data Points	Input From	Output To	Test Point	Characteristics					
				Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
Left Main Wheel Speed		7.0, 8.0	X	Dig.	0-10K PPS	0-1K PPS	0-1500 rpm	±20 rpm	
Right Main Wheel Speed		7.0, 8.0	X	Dig.	0-10K PPS	0-1K PPS	0-1500 rpm	±20 rpm	
Nose Door Pos.		7.0, 8.0	X	On/Off	dc	dc	5V	—	
Air Data Sensor Pos.		5.0 7.0, 8.0	X	Anal	5 Hz	1 Hz	0-36 in.	±1 in.	
Nose Gear Pos.		7.0, 8.0	X	Anal	4 Hz	2 Hz	0-90°	±2°	
Left Main Gear Pos.		7.0, 8.0	X	Anal	4 Hz	2 Hz	0-90°	±2°	
Left Main Gear Door Pos.		7.0, 8.0	X	On/Off	dc	dc	5V	—	
Right Main Gear Pos.		7.0, 8.0	X	Anal	4 Hz	2 Hz	0-90°	±2°	
Right Main Gear Door Pos.		7.0, 8.0	X	On/Off	dc	dc	5V	—	
1.9 LANDING ENGINES & AIR SCOOP DEPLOYMENT									
Landing Phase Initiation Arm	5.0, 6.0 7.0, 8.0			On/Off	dc	dc	24V	—	
Deploy Turbojet Engines Command	5.0, 6.0 7.0, 8.0			On/Off	dc	dc	24V	—	
Turbojet Deployment Controller Temp.			X	Anal	2 Hz	1 Hz	0-200° F	±5° F 0-5V	
Turbojet Deployment Controller Output 1			X	Anal	2 Hz	1 Hz	0-100°	±2° 0-5V	
Turbojet Deployment Controller Output 2			X	Anal	2 Hz	1 Hz	0-110°	±5° 0-5V	
Engine Deployment Actuator Temp.			X	Anal	4 Hz	2 Hz	0-1500° F	±20° F 0-5V	
Engine Deployment Actuator Press.			X	Anal	2 Hz	1 Hz	0-3000 psi	±25 psi 0-5V	
Lower T'jet Doors Actuator Temp.			X	Anal	4 Hz	2 Hz	0-1500° F	±20° F 0-5V	
Lower T'jet Doors Actuator Press.			X	Anal	2 Hz	1 Hz	0-3000 psi	±25 psi 0-5V	
Left Turbojet Pod Pos.			X	Anal	2 Hz	1 Hz	0-100°	±2° 0-5V	
Right Turbojet Pod Pos.			X	Anal	2 Hz	1 Hz	0-100°	±2° 0-5V	
Turbojet Doors Pos.			X	Anal	2 Hz	1 Hz	0-110°	±5° 0-5V	

Data Points	Input From	Output To	Test Point	Characteristics					
				Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
Lt. Air Scoop Inlet Temp.			X	Anal	10 Hz	5 Hz	0-1500° F	±10° F 0-5V	
Lt. Air Scoop Inlet Press.			X	Anal	10 Hz	5 Hz	0-100 psi	±5 psi 0-5V	
Rt. Air Scoop Inlet Temp.			X	Anal	10 Hz	5 Hz	0-1500° F	±10° F 0-5V	
Rt. Air Scoop Inlet Press.			X	Anal	10 Hz	5 Hz	0-100 psi	±5 psi 0-5V	
1.10 VEHICLE CONTROL SURFACES									
Cont. Surface Interlock	5.0, 6.0 7.0, 8.0			On/Off	dc	dc	24V	—	
Right Cont. Surface Command	5.0		X	Anal	25 Hz	10 Hz	0-±45°	±2° 0-5V	
Left Cont. Surface Command	5.0		X	Anal	25 Hz	10 Hz	0-±45°	±2° 0-5V	
Right Elevon Command	5.0		X	Anal	25 Hz	10 Hz	0-±45°	±2° 0-5V	
Left Elevon Command	5.0		X	Anal	25 Hz	10 Hz	0-±45°	±2° 0-5V	
Horizontal Trim Surface Command	5.0		X	Anal	2 Hz	1 Hz	0-60°	±5° 0-5V	
Servo Valve		8.0	X	Anal	4 Hz	1 Hz	0-2000° F	±25° 0-5V	
Servo Valve		8.0	X	Anal	4 Hz	1 Hz	0-3000 psi	±50 psi 0-5V	
Servo Actuator		8.0	X	Anal	4 Hz	1 Hz	0-2000° F	±25° F 0-5V	
Servo Actuator		8.0	X	Anal	4 Hz	1 Hz	0-3000 psi	±50 psi 0-5V	
Right Cont. Surf. Temp.		8.0	X	Anal	10 Hz	5 Hz	0-2500° F	±25° F 0-5V	
Left Cont. Surf. Temp.		8.0	X	Anal	10 Hz	5 Hz	0-2500° F	±25° 0-5V	
Right Elevon Temp.		8.0	X	Anal	10 Hz	5 Hz	0-2500° F	±25° F 0-5V	
Left Elevon Temp.		8.0	X	Anal	10 Hz	5 Hz	0-2500° F	±25° F 0-5V	

Data Points	Input From	Output To	Test Point	Characteristics					
				Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
Horiz Trim Surf. Temp.		8.0	X	Anal	10 Hz	5 Hz	0- 2500°F	±25°F 0-5V	
Right Cont. Surf. Pos.		5.0, 8.0	X	Anal	25 Hz	10 Hz	0-±45°	±2° 0-5V	
Left Cont. Surf. Pos.		5.0, 8.0	X	Anal	25 Hz	10 Hz	0-±45°	±2° 0-5V	
Right Elevon Pos.		5.0, 8.0	X	Anal	25 Hz	10 Hz	0-±45°	±2° 0-5V	
Left Elevon Pos.		5.0, 8.0	X	Anal	25 Hz	10 Hz	0-±45°	±2° 0-5V	
Horiz. Trim Surf. Pos.		5.0, 8.0	X	Anal	2 Hz	1 Hz	0-60°	±5° 0-5V	
1.11 VEHICLE ABORT SENSORS									
Gaseous H_2 Present		7.0, 8.0	X	Disc.	dc	dc	24V	—	
Gaseous O_2 Present		7.0, 8.0	X	Disc.	dc	dc	24V	—	
Fire Present		7.0, 8.0	X	Anal	5 Hz	1 Hz	0- 1500°F	±20°F 0-5V	
Main Engine #1 Firing		7.0, 8.0	X	Anal	20 Hz	5 Hz	0- 4000°F	±100°F 0-5V	
Main Engine #2 Firing		7.0, 8.0	X	Anal	20 Hz	5 Hz	0- 4000°F	±100°F 0-5V	
Main Engine #3 Firing		7.0, 8.0	X	Anal	20 Hz	5 Hz	0- 4000°F	±100°F 0-5V	
Main Engine #4 Firing		7.0, 8.0	X	Anal	20 Hz	5 Hz	0- 4000°F	±100°F 0-5V	
Main Engine #5 Firing		7.0, 8.0	X	Anal	20 Hz	5 Hz	0- 4000°F	±100°F 0-5V	
Main Engine #6 Firing		7.0, 8.0	X	Anal	20 Hz	5 Hz	0- 4000°F	±100°F 0-5V	
Turbojet Engine #1 Firing		7.0, 8.0	X	Anal	10 Hz	5 Hz	0- 2000°F	±50°F 0-5V	
Turbojet Engine #2 Firing		7.0, 8.0	X	Anal	10 Hz	5 Hz	0- 2000°F	±50°F 0-5V	
Turbojet Engine #3 Firing		7.0, 8.0	X	Anal	10 Hz	5 Hz	0- 2000°F	±50°F 0-5V	
Turbojet Engine #4 Firing		7.0, 8.0	X	Anal	10 Hz	5 Hz	0- 2000°F	±50°F 0-5V	
JP-4 Fuel Present		7.0, 8.0	X	Disc.	dc	dc	24V	—	
ECS Toxic Gas Present (#1-#AA)		7.0, 8.0	X	Disc.	dc	dc	24V	—	

2.0 Propulsion

Data Points	Input From	Output To	Test Point	Characteristics					
				Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
2.1 MAIN ENGINES									Per. Eng.
Temp. Fuel Chamb.	2.1	2.2, 8.0	X	Anal	30	10	20-°K	2%	LH_2
Temp. Oxidiz. Chamb.	2.1	2.2, 8.0	X	Anal	30	10	20-°K	2%	LOX
Pos., Nozzle	2.2	7.0	X	dc Volt	Disc.	—	1.5-5.0V	0-5V	
Ignition Voltage	3.0	7.0	X	dc Volt.	On-Off	—		1V	
Ignition Current	3.0	7.0	X	dc Amp	On-Off	—		0.5A	
Press., Fuel Pump	2.2	8.0	X	Anal	30	10		1%	LH_2
Press., Oxi. Pump	2.2	8.0	X	Anal	30	10		1%	LOX
Flow, Fuel	2.2	7.0, 8.0	X	Anal	30	10		0.5%	LH_2
Flow, Oxidizer	2.2	7.0, 8.0	X	Anal	30	10		0.5%	LOX
rpm, Pump	2.1	8.0	X	Anal	15	5		1%	LH_2
rpm, Pump	2.1	8.0	X	Anal	15	5		1%	LOX
Pos., Prop. Cont.	2.1	5.0	X	Anal	30	10	0-5V	2%	
Pos., Preburn Vol.	2.1	5.0	X	Anal	30	10	0-5V	2%	LH_2
Pos., Preburn Vol.	2.1	5.0	X	Anal	30	10	0-5V	2%	LOX
Pos., Main Chamb.	2.1	5.0	X	Anal	30	10	0-5V	2%	LH_2
Pos., Main Chamb.	2.1	5.0	X	Anal	30	10	0-5V	2%	LOX
Temp. Preburn Chamb.	2.1	8.0	X	Anal	15	5		2%	
Temp. Main Burn Skin	2.1	8.0	X	Anal	15	5		3%	
Temp. Heat Exch.	2.1	8.0	X	Anal	15	5		3%	
Temp. Nozz. Cool.	2.1	8.0	X	Anal	15	5		3%	
2.2 MAIN ENG. CONT.									
Eng. Start Signal	5.0	2.1	X	dc Volt.	1 per sec	1 per sec	0-5V	0-5V	Per Eng.
Eng. Nozzle Pos.	5.0	2.1, 7.0	X	dc Volt.	Disc.	Disc.	0-5V	0-5V	Per Eng.
Eng. O/F Ratio	5.0	2.1, 7.0	X	dc Volt.					Per Eng.
Eng. Thrust Cont.	5.0	2.1, 7.0	X	dc Volt.			0-5V	0.5V	Per Eng.
2.3 PROPELLANT SUPPLY									
Level, LH_2 CP (7)	2.3	8.0, 7.0	X	Anal	30	10	0-100%	2%	2 Main/ Tank
Level, LOX CP (7)	2.3	8.0, 7.0	X	Anal	30	10	0-100%	2%	2 Main/ Tank

Data Points	Input From	Output To	Test Point	Characteristics					
				Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
Level, LH_2 OPS (35)	2.3	8.0, 7.0	X	dc Volt.	Disc.	—	0-5V	0.5V	Per Tank
Level LOX OPS (35)	2.3	8.0, 7.0	X	dc Volt.	Disc.	—	0-5V	0.5V	Per Tank
Temp. LH_2 T/C (35)	2.3	8.0, 7.0	X	Anal	15	5	0-5V	0.1V	32-45 °R
Temp. LOX T/C (35)	2.3	8.0, 7.0	X	Anal	15	5	0-5V	0.1V	100-160 °R
Press. LH_2 (16)	2.3	2.4	X	Anal	15	5	0-35 psia	1%	2 Per Tank
Press. LOX (16)	2.3	2.4	X	Anal	15	5	0-35 psia		
2.4 PROPELLANT MGT.									
Volume (16)	2.3	7.0						1%	Per Tank
Press (32)	2.3	7.0						2%	Per Tank
Cont., LH_2	2.4	2.1/2.3	X	dc Volt.	10	3	0-5V	0-5V	Valves
Cont., LOX	2.4	2.1/2.3	X	dc Volt.	10	3	0-5V	0.5V	Valves
2.5 PROPELLANT CONTROL									
Bypass Valve, LH_2	2.4	8.0	X	dc Volt.	On-Off	Disc.	0-5V	0.5V	7-LH_2
Bypass Valve, LOX	2.4	8.0	X	dc Volt.	On-Off	Disc.	0-5V	0.5V	7-LOX
Eng. Manifold Valve	2.4	8.0	X	dc Volt.	On-Off	Disc.	0-5V	0.5V	5-LH_2
Eng. Manifold Valve	2.4	8.0	X	dc Volt.	On-Off	Disc.	0-5V	0.5V	5-LOX
Prop. Manifold Valve	2.4	8.0	X	dc Volt.	On-Off	Disc.	0-5V	0.5V	6-LH_2
Prop. Manifold Valve	2.4	8.0	X	dc Volt.	On-Off	Disc.	0-5V	0.5V	6-LOX
Vent Valves	2.4	8.0	X	dc Volt.	On-Off	Disc.	0-5V	0.5V	4-LH_2
Vent Valves	2.4	8.0	X	dc Volt.	On-Off	Disc.	0-5V	0.5V	4-LOX
2.6 RCS PROPELLANT SUPPLY									
GHe Vent Valve Cmd	5.0, 7.0			On/Off		Disc.			(2)
GHe Vent Valve Pos.		7.0/8.0	X	On/Off		Disc.			(2)
GHe Manifold Valve Cmd	5.0, 7.0			On/Off		Disc.			(4)
GHe Manifold Valve Pos.		7.0/8.0	X	On/Off		Disc.			(4)
Regulator Valve Pos.		8.0	X	Anal	30	5		1%	(4)

Data Points	Input From	Output To	Test Point	Characteristics					
				Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
GHe Temp.		8.0	X	Anal	30	5		2%	6
GHe Press.		7.0, 8.0	X	Anal	30	5		2%	4
N_2O_4 Temp.		8.0	X	Anal	30	5		2%	6
N_2O_4 Press.		7.0, 8.0	X	Anal	30	5		2%	4
N_2O_4 Level		7.0, 8.0	X	Anal	30	10		2%	2
N_2H_3 (CH_3) Temp.		8.0	X	Anal	30	5		2%	6
N_2H_3 (CH_3) Press.		7.0, 8.0	X	Anal	30	5		2%	4
N_2H_3 (CH_3) Level		7.0, 8.0	X	Anal	30	10		2%	2
2.7 RCS PROPELLANT CONTROL									
Disconnect Valve Cmd	5.0, 7.0	8.0	X	On/Off		Disc.			5
Disconnect Valve Pos.		8.0	X	On/Off		Disc.			5
Propellant Vent Valve Cmd	5.0, 7.0	8.0	X	On/Off		Disc.			4
Propellant Vent Valve Pos.		8.0	X	On/Off		Disc.			4
Activate RCS Cmd	5.0, 7.0	8.0	X	On/Off		Disc.			14
Propellant Manifold Valve Pos.		8.0	X	On/Off		Disc.			14
Manifold Press.		8.0	X	Anal	30	5		2%	6
Manifold Temp.		8.0	X	Anal	30	5		2%	6
2.8 RCS THRUSTERS									
Solenoid Valve Cmd	5.0,	8.0	X	On/Off		Disc.			14
NOTE: 14 thrusters/ bipropellant/dual redundancy									
Sol Valve Pos.		8.0	X	On/Off		Disc.			56
Oxidizer Press.		7.0, 8.0	X	Anal	30	5		2%	6
Oxidizer Temp.		8.0	X	Anal	30	5		2%	6
Fuel Press.		7.0, 8.0	X	Anal	30	5		2%	6
Fuel Temp.		8.0	X	Anal	30	5		2%	6
Filter Diff Press.		8.0	X	Anal	30	5		2%	6
2.9 LANDING AID ENGINES (PER ENGINE)									
Ignition Signal	2.10		X	On/Off		Disc.			
Fuel Pump Start	2.10		X	On/Off		Disc.			
Stop Signal	2.10		X	On/Off		Disc.			
Fuel Supply Valve	2.10		X	On/Off		Disc.			
Fuel Cont. Valve	2.10			Anal		5			
Fuel Cont Valve Pos.		2.10	X	Anal		5			

Data Points	Input From	Output To	Test Point	Characteristics					
				Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
Air Cont Valve Cmd	2.10			Anal		5			
Air Cont Valve Cmd		2.10	X	Anal		5			
F/O Ratio Cmd	2.10			Anal		10			
Fuel Pump rpm		8.0	X	Anal		10		0.5%	
Compressor rpm		7.0, 8.0	X	Anal		10		0.5%	
Voltage, FP		8.0	X	Anal		-1			
Voltage, ac		8.0	X	Anal		1			
Eng. Oil Press.		7.0, 8.0	X	Anal		5		2%	
Eng. Lub Oil Press.		7.0, 8.0	X	Anal		5		2%	
Eng. Inlet Press.		7.0, 8.0, 2.10	X	Anal		10		2%	
Eng. Inlet Temp.		8.0, 2.10	X	Anal		10		2%	
Exh. Gas Temp.		8.0	X	Anal		5		2%	
Cont Turbine Temp		7.0, 8.0, 2.10	X	Anal		5		2%	
Qty, Lub Oil		7.0, 8.0	X	Anal		3		5%	
2.10 LANDING AND PROPELLANT SUPPLY/ ENG CONT									
Engine Start	5.0, 7.0		X	On/Off		Disc.			
Engine Stop	5.0, 7.0		X	On/Off		Disc.			
Ignition Cmd (2)		2.9	X	On/Off		Disc.			
Fuel Pump Start (2)		2.9	X	On/Off		Disc.			
Stop Signal		2.9	X	On/Off		Disc.			
Fuel Tank Press. (4)		8.0	X	Anal		5			
Fuel Tank Temp. (4)		8.0	X	Anal		5			
Fuel Tank Level (2)		7.0, 8.0	X	Anal		5			
Fuel Flow Rate (4)		7.0, 8.0	X	Anal		5		1%	
Fuel Tank Disconnect	5.0, 7.0	8.0	X	On/Off		Disc			
Manifold Valve Cmd (4)		8.0	X	On/Off		Disc.			
Manifold Valve Pos. (4)		8.0	X	On/Off		Disc.			
Manifold Press. (3)		7.0, 8.0	X	Anal		5			
Manifold Temp. (3)			X	Anal		5			
F/O Ratio (2)		2.9	X	Anal		10			
Altitude	5.0		X	Dig.		10		1/2%	
Mach No.	5.0		X	Dig.		10		3%	
Air Cont Valve Cmd (2)		2.9, 8.0	X	Anal		5			
Air Cont Valve Pos. (2)	2.9	8.0	X	Anal		5			
Fuel Cont. Valve Cmd (2)		2.9, 8.0	X	Anal		5			

Data Points	Input From	Output To	Test Point	Characteristics					
				Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
Fuel Cont Valve Pos. (2)	2.9	8.0	X	Anal		5			
Eng. Inlet Press. (2)	2.9		X	Anal		10		2%	
Eng. Inlet Temp. (2)	2.9		X	Anal		10		2%	
Eng. Int. Turbine Temp. (2)	2.9		X	Anal		5		2%	
Eng. Thrust Level	5.0		X	Dig.		10		1%	

3.0 Electrical Power

Data Points	Measurement Classification	Type	Range	Resl.	Action	Remarks
1. Voltage — dc Bus	Voltage	A	24-32 V	±1V dc	M D	
2. Current — dc Bus	Current	A	0-125 A	±1A dc	M D	
3. Power — dc Bus	Watts		0-4000 W	±5W	X	Compute Using Functions 1 and 2
4. Voltage — ac Bus	Voltage	A	105-125 V	±2V	M D	
5. Current — ac Bus	Current	A	0-25 A	±1A	M D	
6. Power — ac Bus	Volt-Amps		0-30 kV-A	±5 W	X	Compute Using Functions 4 and 5
FUEL CELL SYSTEM						
FUEL CELL STACK						
1. Voltage	Voltage	A	24-32V	±0.25V	M D	
2. Current	Current	A	10-100A	±1.0A	M D	

NOTE: A = analog, C = control, D = display, M = monitor, and X = compute

Data Points	Measurement Classification	Type	Range	Resl.	Action	Remarks
3. Power Out	kW	—	3 kW	±5W	X	Computed using functions 1 and 2
4. Electrical Energy	kW-h	—	0-400 kW-h	±5 W-h	X	Computed using functions 1 and 2 plus time
5. Inlet Hydrogen Press.	Press.	A	30-50 psia	±1.0 psia	M	$O_2 - H_2 = \pm 1$ psi $O_2 > H_2$
6. Inlet Hydrogen Press.	Press.	A	30-50 psia	±1 psia	M	Alarm if either condition is violated
7. Oxygen Press. Switch	Position	Open-Close		—	M	Closed indication and (5) > 45 psi = alarm
8. Hydrogen Press. Switch	Position	Open-Close		—	M	Closed indication and (6) > 45 psi = alarm
9. Purge Valve O_2	Position	Open-Close		—	C M	Normal purge cycle every 1 kW-h
10. Purge Valve H_2	Position	Open-Close		—	C M	Adjustable from 0.5 to 2 kW-h
11. Oxygen Inlet Valve	Position	Open-Close		—	C[a] M	

a. Not computer-initiated control

NOTE: A = analog, C = control, D = display, M = monitor, and X = compute

Data Points	Measurement Classification	Type	Range	Resl.	Action	Remarks
12. Hydrogen Inlet Valve	Position	Open-Close		—	C[a] M	
13. Stack Temp.	3 Temp.	A	150-210° F	±2° F	M	Alert above 200° F
WATER RECOVERY						
Condenser						
14. By-Pass Valve	Position	Open-Closed		—	M	Command computer initiated only on water tank full indication.
15. Press. Switch	Press.	A	0-5 psia	±0.05 psia	M	Alert if in excess of 4.4 psi.
16. Coolant Temp.	Temp.	A	30-160 °F	±2° F	M	
Water Pump						
17. Full Switch	Position	Open-Closed		—	M	

a. Not computer-initiated control

NOTE: A = analog, C = control, D = display, M = monitor, and X = compute

Data Points	Measurement Classification	Type	Range	Resl.	Action	Remarks
18. Gas Inlet Press.	Press.	A	0-75 psia	±1 psia	M	
19. Press. Relief Valve	Position	Open-Closed		—	M	
20. Gas Stop Valve	Position	Open-Closed		—	M C	Close when condenser by-pass is operated.
21. Gas Vacuum Regulator	Press.	A	0-0.3 psia	±0.05 psia	M	
Moisture Removal						
22. By-Pass Removal Valve	Position	Open-Closed		—	M C	Control not computer initiated
23. Cavity Press.	Press.	A	0-10 psia	±0.05 psia	M	Alert if < 2.5 or > 4.8
24. Electrolyte Concentration	Percent KOH	—		34-41	X	Compute from (24) and (13) alert if <36% of < 40% KOH
THERMAL CONTROL						
25. Cell Heaters	On — Off	—	—	—	C	Not computer initiated (start-up only).

NOTE: A = analog, C = control, D = display, M = monitor, and X = compute

Data Points	Measurement Classification	Type	Range	Resl.	Action	Remarks
26. Helium Press.	Press.	A	0-60 psia	±3 psia	M	
27. Helium Vent	Position	Open-Closed	—	—	M C	
28. Helium Fan #1	rpm		20-28K rpm	±200 rpm	M	If (13) exceeds 200° F turn on (29).
29. Helium Fan #2	rpm		20-28K rpm	±200 rpm	M	(28) & (29) Fail Power Down System
30. Fan #1 Inverter	Current	A	0-3A	±0.5A	M	
31. Fan #2 Inverter	Current	A	0-3A	±0.5A	M	
32. Coolant Pump	On — Off				C	
33. Coolant Pump Inverter	Current	A	0-3A	±0.5A	M	
34. Coolant Pump Press.	Press.	A	10-15 psia	±1 psia	M	
35. Coolant Outlet Temp.	Temp.	A	60°-100° F	±2° F	M	
REACTANT SUPPLY						
36. Oxygen Press.	Press.	A	60-900 psia	±10 psia	M	

NOTE: A = analog, C = control, D = display, M = monitor, and X = compute

Data Points	Measurement Classification	Type	Range	Resl.	Action	Remarks
37. Hydrogen Press.	Press.	A	60-900 psia	±10 psia	M	
38. Oxygen Temp.	Temp.	A	-300°-0° F	±10° F	M	
39. Hydrogen Temp.	Temp.	A	-425°-70° F	±10° F	M	
40. Oxygen Quantity	Volume	—	225-0 lb	±0.2 lb	X	Compute using (36) and (38)
41. Hydrogen Quantity	Volume	—	28-0 lb	±0.1 lb	X	Compute using (37) and (39)
42. Energy Remaining	kW-h	—	400-0 kW-h	±5W-h	X	Compute from (41) as long as (40) > 8x (41)
43. Capacitance Probe O_2	Level	A	100-0 %	±0.1%	M	
44. Capacitance Probe H_2	Level	A	100-0 %	±0.1%	M	
45. Oxygen Vent Valve	Position	Open-Closed		—	M C	

NOTE: A = analog, C = control, D = display, M = monitor, and X = compute

Data Points	Measurement Classification	Type	Range	Resl.	Action	Remarks
46. Oxygen Vent Valve	Position	Open-Closed		—	M C	Command Vent Display warning at design setting of 36 and 37; issue alarm if pressure exceeds excess of Design Settings
47. Oxygen Heater Status	Current	A			M	
48. Hydrogen Heater Status	Current	A			M	
INVERTERS						
1. Voltage	Voltage	A	105-125V	±2V	M	
2. Frequency	Frequency	A	380-420	±2 Hz	M	
3. Power	Watts	A	0-1500W	±5W	X	Compute using functions 1 and 5
4. Temp.	Temp.	A	-20°-85°C	±5°C	M	
5. Current	Current	A	0-50A	±1A	M	

NOTE: A = analog, C = control, D = display, M = monitor, and X = compute

Data Points	Measurement Classification	Type	Range	Resl.	Action	Remarks
6. Voltage — Phase A1	Voltage	A	105-125 V	±2V	M	
7. Voltage — Phase B1	Voltage	A	105-125 V	±2V	M	
8. Voltage — Phase C1	Voltage	A	105-125 V	±2V	M	
9. Voltage — Phase A2	Voltage	A	105-125 V	±2V	M	
10. Voltage — Phase B2	Voltage	A	105-125 V	±2V	M	
11. Voltage — Phase C2	Voltage	A	105-125 V	±2V	M	
AUXILIARY POWER SOURCE						
1. O_2 Supply Valve	Position	Open-Closed			M C	Not computer initiated
2. H_2 Supply Valve	Position	Open-Closed			M C	

NOTE: A = analog, C = control, D = display, M = monitor, and X = compute

Data Points	Measurement Classification	Type	Range	Resl.	Action	Remarks
3. O_2 Temp.	Temp.	A	0-100 °F	±2° F	M	
4. H_2 Temp.	Temp.	A	0-100° F	±2° F	M	
5. Combustor Inlet Press. O_2	Press.	A	300-400 psi	±15 psi	M C	
6. Combustor Inlet Press. H_2	Press.	A	300-400 psi	±15 psi	M C	
7. Combustor Inlet Temp.	Temp.	A	1000 - 1400 °F	±25° F	M	
8. Combustor Outlet Temp.	Temp.	A	1000-1400° F	±25° F	M	
9. Tachometer Status	rpm	A		±0. 5%	M	
10. Turbine Speed	rpm	A	0-70 000 rpm	600 rpm	M C	
11. Lube Oil Temp.	Temp.	A	150-190° F	±5° F	M	
12. Lube Oil Quantity	Position	A		5% of Quant	M	

NOTE: A = analog, C = control, D = display, M = monitor, and X = compute

Data Points	Measurement Classification	Type	Range	Resl.	Action	Remarks
13. Lube Oil Bypass Valve	Position	Open-Closed			M	
14. Regenerator Bypass H_2	Position	Open-Closed			M	
EMERGENCY POWER						
1. Battery Voltage	Voltage	A	27-32	$\pm\frac{1}{2}$V	M	
2. Battery Current	Current	A	0-100	±1A	M	
3. Energy Remaining	Watt-Hours	A	11 000-0 W-h	±50	X	Compute using functions 1 and 2
dc POWER SWITCHING						
1. Status — Internal External	Position	On Off			M	
2. Status — Emergency Power	Position	On Off			M	
3. Status Rect. 1	Position	On Off			M	
4. Status Rect. 2	Position	On Off			M	

NOTE: A = analog, C = control, D = display, M = Monitor, and X = compute

Data Points	Measurement Classification	Type	Range	Resl.	Action	Remarks
ac POWER SWITCHING						
1. Status — Internal / External	Position	On / Off			M	
2. Status — Interlock	Position	On / Off			M	
3. Inverter No. 1	Position	On / Off			M	
4. Inverter No. 2	Position	On / Off			M	
POWER CONTROL						
1. Status Int./Ext. — dc	Position	On / Off			M	
2. Status Emergency	Position	On / Off			M	
3. Overload Condition — dc	Position	On / Off			M / C	
4. Status Inv. 1 or 2	Position	On / Off			M	

NOTE: A = analog, C = control, D = display, M = monitor, and X = compute

Data Points	Measurement Classification	Type	Range	Resl.	Action	Remarks
5. Status Alternator 1, 2, or 3	Position	On Off				
6. Status Int./Ext. — ac	Position	On Off			M	
7. Overload Condition — ac	Position	On Off			M C	
8. Power Computation — dc	Watts	A	0-10 kW	±200 W	M	
9. Power Computation — ac	Watts	A	0-1.5 kV-A	±50 V-A	M	

NOTE: A = analog, C = control, D = display, M = monitor, and X = compute

4.0 Environmental Control/Life Support

Data Points	Input From	Output To	Test Point	Characteristics					
				Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
4.1 GAS SUPPLY									
N_2 Tank #1 Temp. Sensor		8.0	X	Anal	≈dc		-300°-100°	±10°	
N_2 Tank #2 Temp. Sensor		8.0	X	Anal	≈dc		-300°-100°	±10°	
N_2 Tank #1 Press. Sensor		8.0,7.0	X	Anal	≈dc		0-1500 psi	±30 psi	
N_2 Tank #2 Press. Sensor		8.0,7.0	X	Anal	≈dc		0-1500 psi	±30 psi	
N_2 Tank #1 Quantity		8.0,7.0	X	Anal	≈dc		0-100%	±1%	
N_2 Tank #2 Quantity		8.0,7.0	X	Anal	≈dc		0-100%	±1%	
N_2 Tank #1 Heater		8.0	X	On/Off	400 Hz	Once every 30 sec	0 or 115V	±20V	
N_2 Tank #2 Heater		8.0	X	On/Off	400 Hz		0 or 115V	±20V	
Switch Empty N_2 to Full N_2	7.0		X	On/Off	400 Hz		0 or 115V	±20V	
O_2 Tank #1 Temp. Sensor		8.0	X	Anal	≈dc		-300°-100°	±10°	
O_2 Tank #2 Temp. Sensor		8.0	X	Anal	≈dc		-300°-100°	±10°	
O_2 Tank #1 Press. Sensor		8.0,7.0	X	Anal	≈dc		0-1500 psi		
O_2 Tank #2 Press. Sensor		8.0,7.0	X	Anal	≈dc		0-1500 psi		
O_2 Tank #1 Quantity		8.0,7.0	X	Anal	≈dc		0-100%		
O_2 Tank #2 Quantity		8.0,7.0	X	Anal	≈dc		0-100%		
O_2 Tank #1 Heater		8.0	X	On/Off	400 Hz	Once every 30 sec	0 or 115V	±20V	
O_2 Tank #2 Heater		8.0	X	On/Off	400 Hz		0 or 115V	±20V	
Switch Empty O_2 to Full O_2 Tank	7.0,8.0		X	On/Off	400 Hz		0 or 115V	±20V	
4.2 ATMOSPHERE SUPPLY CONTROL									
Cabin Press. Sensor		8.0,7.0	X	Anal	10#/sec	5#/sec	0-17 lb	0.1 lb	
Oxygen Partial Press.		8.0	X	Anal	10#/sec	5#/sec	0-10 lb	0.1 lb	
External Press. Sensor		8.0	X	Anal	10#/sec	5#/sec	0-17 lb	0.1 lb	

Data Points	Input From	Output To	Test Point	Characteristics					
				Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
Cabin ΔP Sensor		8.0	X	On/Off	—	—	0-17 lb	1 lb/sec closes SW	
O_2 Intermediate Press. Sensor		8.0	X	Anal	10#/ sec	5#/ sec	0-150 lb	±1 lb	
N_2 Intermediate Press. Sensor		8.0	X	Anal	10#/ sec	5#/ sec	0-150 lb	±1 lb	
O_2 Flow Rate Sensor		8.0, 7.0	X	Anal	10#/ sec	5#/ sec	0.2-1 lb/hr	±0.02 lb/hr	
N_2 Flow Rate Sensor		8.0, 7.0	X	Anal	10#/ sec	5#/ sec	0.2-1 lb/hr	±0.02 lb/hr	
N_2 Release Valve Cont.	7.0	8.0	X	Anal	400 Hz		0-115V	±1% Motor Cont.	
O_2 Release Cont.	7.0	8.0	X	Anal	400 Hz		0-115V	±1% Motor Cont.	
Cabin Air Release-Admit Valve Cont.	7.0		X	On/Off	400 Hz		0 or 115V		
Emergency O_2 Release Valve	7.0		X	On/Off	400 Hz		0 or 115V		
Emergency O_2 Source Valve	7.0		X	On/Off	400 Hz		0 or 115V		
Cabin Press. High Cont.	7.0		X	On/Off	dc		0 or 5V		
Oxygen Low Press. Cont.	7.0		X	On/Off	dc		0 or 5V		
Oxygen Flow Rate High Cont.	7.0		X	On/Off	dc		0 or 5V		
Nitrogen Flow Rate High Cont.	7.0		X	On/Off	dc		0 or 5V		
Suit O_2 Temp Sensor		8.0, 7.0	X	Anal	τ= 5 sec	τ= 10 sec	0-100°F	±5° 0-5V	
Suit O_2 Temp Ref.		8.0	X	Anal	Manually Variable		20°-90°F	±2° 0-5V	
Suit O_2 Evap. Back Press. Sensor		8.0	X	Anal	τ= 1/2 sec	τ= 1/2 sec	0-15 psi	±0.1 psi	
Back Press. Wetness Sensor		8.0	X	Anal	τ= 1/2 sec	τ= 1/2 sec	0-100%	±2%	
Suit #1 Press. Sensor		8.0, 7.0	X	Anal	τ= 1 sec		0-10 psi	±0.1 psi 0-5V	
Suit #2 Press. Sensor		8.0, 7.0	X	Anal	τ= 1 sec		0-10 psi	±0.1 psi 0-5V	
Suit #1 Airflow Sensor		8.0	X	On/Off			0 or 5V	Produce Signal at Min. Airflow	
Suit #2 Airflow Sensor		8.0	X	On/Off			0 or 5V		
Suit Press. Sensor		8.0	X	Anal	τ= 0.1 sec		0-5 psi	±0.1 psi	
Suit Compressor — Motor #1 / Motor #2		8.0	X	On/Off	400 Hz		0 or 115V		

Data Points	Input From	Output To	Test Point	Characteristics: Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
Cabin Air-Suit Loop Valve	7.0	8.0	X	On/Off	400 Hz		0 or 115V		
Suit Loop-Cabin Air Valve	7.0	8.0	X	On/Off	400 Hz		0 or 115V		
O_2 Release Valve	7.0	8.0	X	On/Off	400 Hz		0 or 115V		
4.3 CREW WATER CONTROL									
CO_2 Partial Press. Sensor		8.0	X	Anal	≈dc		0-10mm Hg	0.1mm Hg; 0-5V	
Humidity Sensor		8.0	X	Anal	≈dc		0-100%	±5% 0-5V	
Water Accumulator Sensor		8.0	X	Anal	≈dc		0-100%	±5% 0-5V	
Water Separator Cont.	7.0		X	On/Off	400 Hz	Every 10 sec or more		0 or 115V	
LiOH Coolant Heating Cont. Valves	7.0		X	On/Off	400 Hz			0 or 115V	
4.4 CABIN TEMPERATURE CONTROL									
Cabin Temp Sensor		8.0, 7.0	X	Anal	τ= 30 sec	5 min	0-100°F	±1°F 0-5V	
Cabin Temp Reference		8.0	X	Anal	Manual	—	40°F-100°F	±1°F 0-5V	
Cabin Temp Cont.	7.0		X	On/Off	400 Hz		0 or 115V		
Cabin Fan	7.0	8.0	X	On/Off	400 Hz		0 or 115V		
Heat Exch. Bypass Valve	7.0		X	On/Off	400 Hz	Once each 5 sec	0 or 115V		
Back Press. Release Valve	7.0		X	On/Off	400 Hz	Once each sec	0 or 115V		
Water Release Valve	7.0		X	On/Off	400 Hz	Once each sec	0 or 115V		
4.5 COOLANT CONTROL									
Suit Coolant Temp Sensor		8.0	X	Anal	1°/sec	1°/sec	40°-100°	±2° 0-5V	
Suit Coolant Temp Ref		8.0	X	Anal	—	—	40°-100°	±2° 0-5V	
Suit Water Boil Press. Sensor		8.0	X	Anal	1p every 10 sec	1p every 30 sec	0-10 psi	±0.1 psi 0-5V	
Suit Coolant Bypass Valve	7.0	8.0	X	On/Off	400 Hz	—	0 or 115V	—	

Data Points	Input From	Output To	Test Point	Characteristics: Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
Suit Coolant Pump Motor	7.0	8.0	X	On/Off	400 Hz	—	0 or 115V	—	
Coolant Temp Sensor		8.0	X	Anal	τ= 10 sec	τ= 30 sec	-50° 100°F	±1°F 0-5V	
Coolant Temp Reference		8.0	X	Anal	dc				Fixed Ref.
Coolant Reserve Quantity		8.0	X	Anal	dc		0-100%	±2% 0-5V	
Freon Water Boil Press. Sensor		8.0	X	Anal	τ=10 sec	τ=30 sec	0-16 psi	±0.1 psi 0-5V	
Coolant Pump	7.0		X	Off/On	400 Hz			0 or 115V	
Freon Water Boil Press. Cont.	7.0		X	Off/On	400 Hz			0 or 115V	
Water Release Valve	7.0		X	Off/On	400 Hz			0 or 115V	
Glycol Coolant Temp Outlet Sensor		8.0	X	Anal	τ=30 sec	τ=60 sec	-50°-150°	±2° 0-5V	
Glycol Coolant Temp Ref.		8.0	X	—			0-5V		Variable Ref.
Glycol Water Boil Press. Sensor		8.0	X	Anal	τ=1 sec	τ=10 sec	0-17 psi	±0.1 psi 0-5V	
Glycol Coolant Press. Sensor		8.0	X	Anal	τ=1 sec	τ=10 sec	0-150 psi	±2 psi 0-5V	
Clycol Coolant Temp Inlet Sensor		8.0	X	Anal	τ=30 sec	τ=60 sec	-50°-150°	±2° 0-5V	
Waste Tank Quantity		8.0	X	Anal	τ=60 sec	τ=60 sec	0-100%	±2% 0-5V	
Intercool Bypass Valve		8.0	X	On/Off	400 Hz		0 or 115V		
Coolant Pump Motor #1 #2	7.0	8.0	X	On/Off	400 Hz		0 or 115V		
Glycol Coolant Press. Cont.	7.0		X	On/Off	400 Hz		0 or 115V		
Glycol Coolant Press. High & Low		8.0	X	On/Off	—		0 or 5V		
Coolant Press. Cont. Valve	7.0		X	On/Off	400 Hz		0 or 115V		

5.0 Guidance, Navigation, and Control

Data Points	Input From	Output To	Test Point	Characteristics					
				Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
Att. Increments ($+\Delta\theta$)		5.0	X	Dig.	1800	1800	1 Bit		
($-\Delta\theta$)		5.0	X	Dig.	1800	1800	1 Bit		
($+\Delta\psi$)		5.0	X	Dig.	1800	1800	1 Bit		
($-\Delta\psi$)		5.0	X	Dig.	1800	1800	1 Bit		
($+\Delta\phi$)		5.0	X	Dig.	1800	1800	1 Bit		
($-\Delta\phi$)		5.0	X	Dig.	1800	1800	1 Bit		
Scale Factor (Hi-Lo)		5.0	X	Dig.	1800	1800	1 Bit		Parallel Word @ CI
Vel. Increments ($+\Delta Vx$)	5.0	5.0	X	Dig.	1800	1800	1 Bit		
($-\Delta Vx$)		5.0	X	Dig.	1800	1800	1 Bit		
($+\Delta Vy$)		5.0	X	Dig.	1800	1800	1 Bit		
($-\Delta Vy$)		5.0	X	Dig.	1800	1800	1 Bit		
($+\Delta Vz$)		5.0	X	Dig.	1800	1800	1 Bit		
($-\Delta Vz$)		5.0	X	Dig.	1800	1800	1 Bit		
Comptr. Interrupt (CI)		5.0	X	Pulse	1800	1800	10μ sec		R/T
IMU Pwr	3.0		X	Disc.	dc	—	—	—	28V
Local Vert. Errors (θs)		5.0	X	Dig.	1 Hz	1 Hz	±10 Dig.	±0.01 Dig.	
(θs)		5.0	X	Dig.	1 Hz	1 Hz	±10 Dig.	±0.01 Dig.	
Hor. Sens. Pwr	3.0		X	Disc.	dc	—	—	—	28V
Star Sensor Pwr	3.0		X	Disc.	dc	—	—	—	28V
Boresight Error (AZ)		5.0	X	Dig.	5 Hz	15 Hz	16 min	6 sec	
(EL)		5.0	X	Dig.	5 Hz	15 Hz	16 min	6 sec	
Mode (Search/Track)		5.0	X	Disc.	$\frac{0.3}{\text{min}}$	$\frac{0.3}{\text{min}}$	1 Bit		Avg
Altitude (h)		5.0	X	Dig.	5 Hz	1 Hz	350 nm	0.02	
Alt. Rate ($\dot{h}$)		5.0	X	Dig.	5 Hz	1 Hz	10 $\frac{\text{nm}}{\text{min}}$	0.1 $\frac{\text{nm}}{\text{min}}$	
Radar Alt. Pwr	3.0		X	Disc.	dc	—	—	—	28V
Altitude (h)		5.0	X	Dig.	2.5 kHz	5 Hz	40K	2 ft	
Alt. Rate (h)		5.0		Dig.	2 Hz	5 Hz	±500	1 fps	
FM-CW Radar Alt. Pwr	3.0		X	Disc.	dc	—	—	—	28V
Tacan Pwr	3.0		X	Disc.	dc	—	—	—	28V
Slant Range		5.0	X	Dig.	30 Hz	2 Hz	300 nm	0.1 nm	
Bearing		5.0	X	Dig.	15 Hz	2 Hz	360 Dig.	0.25 Dig.	

Data Points	Input From	Output To	Test Point	Characteristics					
				Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
Comptr Pwr	3.0		X	Disc.	dc	—	—	—	28V
ILS Localizer Pwr	3.0		X	Disc.	dc	—	—	—	28V
Azimuth Deviation		5.0	X	Dig.	9 Hz	5 Hz	±2.5 Dig.	0.05°	
ILS Glide Slope Pwr	3.0		X	Disc.	dc	—	—	—	28V
Elev. Deviation		5.0	X	Dig.	9 Hz	5 Hz	2.7+0.3 -0.5	0.05°	
ILS Marker RCVR Pwr	3.0	5.0	X	Disc.	dc	—	—	—	28V
Range Markers		5.0	X	Pulse	3/ Approach		1 Bit	—	Range
Rend. Radar Pwr	3.0		X	Disc.	dc	—	—	—	28V
Range (R)		5.0	X	Dig.	100 Hz	2 Hz	25 nm	0.5 ft	
Closure Rate (R)		5.0	X	Dig.	100 Hz	2 Hz	350	0.5 fps	
Boresight Error (AZ)		5.0	X	Dig.	1 Hz	2 Hz	±45°	0.02°	
(EL)		5.0	X	Dig.	1 Hz	2 Hz	±45°	0.02°	
Air Data Cmptr Pwr	3.0		X	Disc.	dc	—	—	—	28V
Airspeed		5.0	X	Dig.	25 Hz	50 Hz	800 KTS	4 KTS	
Angle of Attack		5.0	X	Dig.	25 Hz	50 Hz	±30°	0.1%	
Side Slip Angle		5.0	X	Dig.	25 Hz	50 Hz	±30°	0.1%	
Total Press.		5.0	X	Anal	200 Hz	200 Hz	0-2 AT	0.1%	
Static Press.		5.0	X	Anal	200 Hz	200 Hz	0-1 AT	0.1%	
Diff. Press.		5.0	X	Anal	200 Hz	200 Hz	0-2 AT	0.1%	
Diff. Press.		5.0	X	Anal	200 Hz	200 Hz	0-2 AT	0.1%	
Temp.		5.0	X	Anal	1 Hz	1 Hz	200°	0.1%	
Pitch Rate Cmd Man.	7.0		X	Dig.	1 Hz	1 Hz	30 Bits		
Yaw Rate Cmd Man.	7.0		X	Dig.	1 Hz	1 Hz	30 Bits		
Roll Rate Cmd Man.	7.0		X	Dig.	1 Hz	1 Hz	30 Bits		
Pitch Att Cmd Man.	7.0		X	Dig.	1 Hz	1 Hz	30 Bits		
Yaw Att Cmd Man.	7.0		X	Dig.	1 Hz	1 Hz	30 Bits		
Roll Att Cmd Man.	7.0		X	Dig.	1 Hz	1 Hz	30 Bits		Computer Words
Pitch Rate Error		5.0	X	Dig.	50 Hz	50 Hz	30 Bits		
Yaw Rate Error		5.0	X	Dig.	50 Hz	50 Hz	30 Bits		
Roll Rate Error		5.0	X	Dig.	50 Hz	50 Hz	30 Bits		
Accel. Cmd.		5.0	X	Dig.	50 Hz	50 Hz	30 Bits		
Angle of Attach Cmd.		5.0	X	Dig.	50 Hz	50 Hz	30 Bits		
Side Slip Angle Cmd.		5.0	X	Dig.	50 Hz	50 Hz	30 Bits		

Data Points	Input From	Output To	Test Point	Characteristics					
				Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
Bank Angle Cmd		5.0	X	Dig.	50 Hz	50 Hz	30 Bits		
Angle of Attack		5.0	X	Dig.	50 Hz	50 Hz	30 Bits		Computer
Slip Angle		5.0	X	Dig.	50 Hz	50 Hz	30 Bits		Words
Bank Angle		5.0	X	Dig.	50 Hz	50 Hz	30 Bits		
Displays (23 Required)		7.0							

Data Points	Input From	Output To	Test Point	Characteristics: Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
Displays (43 Required)		7.0							
Cmptr Pwr	3.0		X	Disc.	dc	—	—	—	28V
Pitch Rate Cmd	5.0		X	Dig.	50 Hz	50 Hz	30 Bits		
Yaw Rate Cmd	5.0		X	Dig.	50 Hz	50 Hz	30 Bits		
Roll Rate Cmd	5.0		X	Dig.	50 Hz	50 Hz	30 Bits		
Accel. Cmd	5.0		X	Dig.	50 Hz	50 Hz	30 Bits		
Angle of Attack Cmd	5.0		X	Dig.	50 Hz	50 Hz	30 Bits		
Side Slip Angle Cmd	5.0		X	Dig.	50 Hz	50 Hz	30 Bits		Cmptr
Bank Angle Cmd	5.0		X	Dig.	50 Hz	50 Hz	30 Bits		Word
Angle of Attack	5.0		X	Dig.	50 Hz	50 Hz	30 Bits		
Side Slip Angle	5.0		X	Dig.	50 Hz	50 Hz	30 Bits		
Angle of Attack	5.0		X	Dig.	50 Hz	50 Hz	30 Bits		
Side Slip Angle	5.0		X	Dig.	50 Hz	50 Hz	30 Bits		
Cont.Config.	5.0		X	Dig.	50 Hz	50 Hz	30 Bits		
TVC Pitch Man. Cmd	7.0	5.0	X	Anal	1 Hz	3 Hz	±10V	0.1%	
TVC Yaw Man. Cmd	7.0	5.0	X	Anal	1 Hz	3 Hz	±10 V	0.1%	
TVC Roll Man. Cmd	7.0	5.0	X	Anal	1 Hz	3 Hz	±10V	0.1%	
TVC Throttle Man. Cmd	7.0	5.0	X	Anal	1 Hz	3 Hz	±10V	0.1%	
RCS Pitch Man. Cmd	7.0	5.0	X	Anal	1 Hz	3 Hz	±10V	0.1%	
RCS Yaw Man. Cmd	7.0	5.0	X	Anal	1 Hz	3 Hz	±10V	0.1%	
RCS Roll Man. Cmd	7.0	5.0	X	Anal	1 Hz	3 Hz	±10V	0.1%	
RCS Roll Man. Cmd	7.0	5.0	X	Anal	1 Hz	3 Hz	±10V	0.1%	
RCS Y Man. Cmd	7.0	5.0	X	Anal	1 Hz	3 Hz	±10V	0.1%	
RCS Z Man. Cmd	7.0	5.0	X	Anal	1 Hz	3 Hz	±10V	0.1%	
Cont Surf Pitch Man. Cmd	7.0	5.0	X	Anal	1 Hz	3 Hz	±10V	0.1%	
Cont Surf Roll Man. Cmd	7.0	5.0	X	Anal	1 Hz	3 Hz	±10V	0.1%	
Cont Surf Yaw Man. Cmd	7.0	5.0	X	Anal	1 Hz	3 Hz	±10V	0.1%	
Cont Surf α Trim Man. Cmd	7.0	5.0	X	Anal	1 Hz	3 Hz	±10V	0.1%	
Cont Surf P Trim Man. Cmd	7.0	5.0	X	Anal	1 Hz	3 Hz	±10V	0.1%	
Turbo Throttle Man. Cmd	7.0	5.0	X	Anal	1 Hz	3 Hz	±10V	0.1%	

Data Points	Input From	Output To	Test Point	Characteristics					
				Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
TVC Cmd (δ1P)	5.0	5.0	X	Anal	50 Hz	50 Hz	±10V	0.1%	
TVC Cmd (δ2P)	5.0	5.0	X	Anal	50 Hz	50 Hz	±10V	0.1%	
TVC Cmd (δ3P)	5.0	5.0	X	Anal	50 Hz	50 Hz	±10V	0.1%	
TVC Cmd (δ4P)	5.0	5.0	X	Anal	50 Hz	50 Hz	±10V	0.1%	
TVC Cmd (δ5P)	5.0	5.0	X	Anal	50 Hz	50 Hz	±10V	0.1%	
TVC Cmd (δ1Y)	5.0	5.0	X	Anal	50 Hz	50 Hz	±10V	0.1%	
TVC Cmd (δ2Y)	5.0	5.0	X	Anal	50 Hz	50 Hz	±10V	0.1%	
TVC Cmd (δ3Y)	5.0	5.0	X	Anal	50 Hz	50 Hz	±10V	0.1%	
TVC Cmd (δ4Y)	5.0	5.0	X	Anal	50 Hz	50 Hz	±10V	0.1%	
TVC Cmd (δ5Y)	5.0	5.0	X	Anal	50 Hz	50 Hz	±10V	0.1%	
TVC ACC Cmd (δ1T)	5.0	5.0	X	Anal	50 Hz	50 Hz	±10V	0.1%	
TVC ACC Cmd (δ2T)	5.0	5.0	X	Anal	50 Hz	50 Hz	±10V	0.1%	
TVC ACC Cmd (δ3T)	5.0	5.0	X	Anal	50 Hz	50 Hz	±10V	0.1%	
TVC ACC Cmd (δ4T)	5.0	5.0	X	Anal	50 Hz	50 Hz	±10V	0.1%	
TVC ACC Cmd (δ5T)	5.0	5.0	X	Anal	50 Hz	50 Hz	±10V	0.1%	
RCS Cmd (δ1)	5.0	5.0	X	Disc.	Avg 0.01 Hz	R/T	40 K	0.001 sec	
RCS Cmd (δ2)	5.0	5.0	X	Disc.	Avg 0.01 Hz	R/T	40 K	0.001 sec	
RCS Cmd (δ3)	5.0	5.0	X	Disc.	Avg 0.01 Hz	R/T	40 K	0.001 sec	
RCS Cmd (δ4)	5.0	5.0	X	Disc.	Avg 0.01 Hz	R/T	40 K	0.001 sec	
RCS Cmd (δ5)	5.0	5.0	X	Disc.	Avg 0.01 Hz	R/T	40 K	0.001 sec	
RCS Cmd (δ6)	5.0	5.0	X	Disc.	Avg 0.01 Hz	R/T	40 K	0.001 sec	
RCS Cmd (δ7)	5.0	5.0	X	Disc.	Avg 0.01 Hz	R/T	40 K	0.001 sec	
RCS Cmd (δ8)	5.0	5.0	X	Disc.	Avg 0.01 Hz	R/T	40 K	0.001 sec	
RCS Cmd (δ9)	5.0	5.0	X	Disc.	Avg 0.01 Hz	R/T	40 K	0.001 sec	
RCS Cmd (δ10)	5.0	5.0	X	Disc.	Avg 0.01 Hz	R/T	40 K	0.001 sec	
RCS Cmd (δ11)	5.0	5.0	X	Disc.	Avg 0.01 Hz	R/T	40 K	0.001 sec	
RCS Cmd (δ12)	5.0	5.0	X	Disc.	Avg 0.01 Hz	R/T	40 K	0.001 sec	
RCS Cmd (δ13)	5.0	5.0	X	Disc.	Avg 0.01 Hz	R/T	40 K	0.001 sec	
RCS Cmd (δ14)	5.0	5.0	X	Disc.	Avg 0.01 Hz	R/T	40 K	0.001 sec	

Data Points	Input From	Output To	Test Point	Characteristics					
				Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
Turbo Throttle Cmd (δ1)	5.0	5.0	X	Anal	50 Hz	50 Hz	±10V	0.1%	
Turbo Throttle Cmd (δ2)	5.0	5.0	X	Anal	50 Hz	50 Hz	±10V	0.1%	
Lt. Elevon Cmd	5.0	5.0	X	Anal	50 Hz	50 Hz	±10V	0.1%	
Rt. Elevon Cmd	5.0	5.0	X	Anal	50 Hz	50 Hz			
Trim Elevon Cmd	5.0	5.0	X	Anal	50 Hz	50 Hz			
Lt. Rudder Cmd	5.0	5.0	X	Anal	50 Hz	50 Hz			
Lt. Trim Rudder	5.0	5.0	X	Anal	50 Hz	50 Hz			
Rt. Rudder Cmd	5.0	5.0	X	Anal	50 Hz	50 Hz			
Rt. Trim Rudder Cmd	5.0	5.0	X	Anal	50 Hz	50 Hz	±10V	0.1%	
Gimbal Pos.	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Gimbal Pos.	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Gimbal Pos.	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Gimbal Pos.	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Gimbal Pos.	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Gimbal Pos.	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Gimbal Pos.	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Gimbal Pos.	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Gimbal Pos.	2.2	5.0	X	An	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Gimbal Pos.	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Gimbal Rate	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Gimbal Rate	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Gimbal Rate	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Gimbal Rate	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Gimbal Rate	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Gimbal Rate	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Gimbal Rate	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Gimbal Rate	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier

Data Points	Input From	Output To	Test Point	Characteristics					
				Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
Gimbal Rate	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Gimbal Rate	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Pri Eng Thr. Pos	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Pri Eng Thr. Pos	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Pri Eng Thr. Pos	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Pri Eng Thr. Pos	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Pri Eng Thr. Pos	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Turbo Throttle Pos.	2.9	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Turbo Throttle Pos.	2.9	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Cont Surf. Pos.	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Cont Surf. Pos.	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Cont Surf. Pos.	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Cont Surf. Pos.	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Cont Surf. Pos.	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Cont Surf. Pos.	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Cont Surf. Pos.	2.2	5.0	X	Anal	400 Hz	3 Hz	±10V rms	0.1%	Modulated carrier
Gimbal Servo Error		2.2	X	Anal	30 Hz	30 Hz	±10 ma	1%	Diff. Current Source
Gimbal Servo Error		2.2	X	Anal	30 Hz	30 Hz	±10 ma	1%	Diff. Current Source
Gimbal Servo Error		2.2	X	Anal	30 Hz	30 Hz	±10 ma	1%	Diff. Current Source
Gimbal Servo Error		2.2	X	Anal	30 Hz	30 Hz	±10 ma	1%	Diff. Current Source
Gimbal Servo Error		2.2	X	Anal	30 Hz	30 Hz	±10 ma	1%	Diff. Current Source
Gimbal Servo Error		2.2	X	Anal	30 Hz	30 Hz	±10 ma	1%	Diff. Current Source

Data Points	Input From	Output To	Test Point	Characteristics					
				Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
Gimbal Servo Error		2.2	X	Anal	30 Hz	30 Hz	±10 ma	1%	Diff. Current Source
Gimbal Servo Error		2.2	X	Anal	30 Hz	30 Hz	±10 ma	1%	Diff. Current Source
Gimbal Servo Error		2.2	X	Anal	30 Hz	30 Hz	±10 ma	1%	Diff. Current Source
Gimbal Servo Error		2.2	X	Anal	30 Hz	30 Hz	±10 ma	1%	Diff. Current Source
Pri Eng Thr Servo Error		2.2	X	Anal	30 Hz	30 Hz	2.5 msec	1%	Bipolar Voltage Source
Pri Eng Thr Servo Error		2.2	X	Anal	30 Hz	30 Hz	2.5 msec	1%	Bipolar Voltage Source
Tri Eng Thr Servo Error		2.2	X	Anal	30 Hz	30 Hz	2.5 msec	1%	Bipolar Voltage Source
Pri Eng Thr Servo Error		2.2	X	Anal	30 Hz	30 Hz	2.5 msec	1%	Bipolar Voltage Source
Pri Eng Thr Servo Error		2.2	X	Anal	30 Hz	30 Hz	2.5 msec	1%	Bipolar Voltage Source
Turbo Thr Servo Error		2.9	X	Anal	30 Hz	30 Hz	2.5 msec	1%	Bipolar Voltage Source
Turbo Thr Servo Error		2.9	X	Anal	30 Hz	30 Hz	2.5 msec	1%	Bipolar Voltage Source
Cont Surf Servo Error		1.10	X	Anal	30 Hz	30 Hz	±10 ma	1%	Diff. Current Source
Cont Surf Servo Error		1.10	X	Anal	30 Hz	30 Hz	±10 ma	1%	Diff. Current Source
Cont Surf Servo Error		1.10	X	Anal	30 Hz	30 Hz	±10 ma	1%	Diff. Current Source
Cont Surf Servo Error		1.10	X	Anal	30 Hz	30 Hz	±10 ma	1%	Diff. Current Source
Cont Surf Servo Error		1.10	X	Anal	30 Hz	30 Hz	±10 ma	1%	Diff. Current Source
Cont Surf Servo Error		1.10	X	Anal	30 Hz	30 Hz	±10 ma	1%	Diff. Current Source
Cont Surf Servo Error		1.10	X	Anal	30 Hz	30 Hz	±10 ma	1%	Diff. Current Source

Data Points	Input From	Output To	Test Point	Characteristics					
				Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
RCS Valve On (1)		2.8	X	Disc.	Avg 0.01 sec	R/T	28V 1A	0.001 sec	On/Off
RCS Valve On (2)	5.0	2.8	X	Disc.	Avg 0.01 sec	R/T	28V 1A	0.001 sec	
RCS Valve On (3)	5.0	2.8	X	Disc.	Avg 0.01 sec	R/T	28V 1A	0.001 sec	
RCS Valve On (4)	5.0	2.8	X	Disc.	Avg 0.01 sec	R/T	28V 1A	0.001 sec	
RCS Valve On (5)	5.0	2.8	X	Disc.	Avg 0.01 sec	R/T	28V 1A	0.001 sec	
RCS Valve On (6)	5.0	2.8	X	Disc.	Avg 0.01 sec	R/T	28V 1A	0.001 sec	
RCS Valve On (7)	5.0	2.8	X	Disc.	Avg 0.01 sec	R/T	28V 1A	0.001 sec	
RCS Valve On (8)	5.0	2.8	X	Disc.	Avg 0.01 sec	R/T	28V 1A	0.001 sec	
RCS Valve On (9)	5.0	2.8	X	Disc.	Avg 0.01 sec	R/T	28V 1A	0.001 sec	
RCS Valve On (10)	5.0	2.8	X	Disc.	Avg 0.01 sec	R/T	28V 1A	0.001 sec	
RCS Valve On (11)	5.0	2.8	X	Disc.	Avg 0.01 sec	R/T	28V 1A	0.001 sec	
RCS Valve On (12)	5.0	2.8	X	Disc.	Avg 0.01 sec	R/T	28V 1A	0.001 sec	
RCS Valve On (13)	5.0	2.8	X	Disc.	Avg 0.01 sec	R/T	28V 1A	0.001 sec	
RCS Valve On (14)	5.0	2.8	X	Disc.	Avg 0.01 sec	R/T	28V 1A	0.001 sec	
Rt. Aft Elec Pwr	3.0		X	Disc.	dc	R/T	28V 1A	0.001 sec	28V
Lt Aft	3.0		X	Disc.	dc				
Fwd	3.0		X	Disc.	dc				
TVC Drive Elec Pwr	3.0		X	Disc.	dc	R/T	28V 1A	0.001 sec	28V
TVC Drive Elec Pwr	3.0		X	Disc.	400 Hz	R/T	28V 1A	0.001	26V rms
Rt CS Elec Pwr	3.0		X	Disc.	dc				28V

Data Points	Input From	Output To	Test Point	Characteristics: Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
Rt CS Elect Pwr	3.0		X	Disc.	400				26V
Lt CS Elect Pwr	3.0		X	Disc.	dc				28V
Lt CS Elect Pwr	3.0		X	Disc.	400				26V
Rt Turbo Elect Pwr	3.0		X	Disc.	dc				28V
Rt Turbo Elect Pwr	3.0		X	Disc.	400				26V
Lt Turbo Elect Pwr	3.0		X	Disc.	dc				28V
Lt Turbo Elect Pwr	3.0		X	Disc.	400				26V
Temp Sensor Pwr	3.0		X	Disc.	dc	R/T	$\frac{28V}{1A}$	0.001 sec	28V
Avg. Skin Temp		5.0	X	Dig	8°/sec	4 Hz	2500°	0.1%	
Differential Skin Temp		5.0	X	Dig.	8°/sec	4 Hz	±500°	0.1%	
Temp Sensor 1		5.0	X	Anal	8°/sec	4 Hz	2500°	0.1%	
Temp Sensor 2		5.0	X	Anal	8°/sec	4 Hz	2500°	0.1%	
Temp Sensor 3		5.0	X	Anal	8°/sec	4 Hz	2500°	0.1%	
Temp Sensor 4		5.0	X	Anal	8°/sec	4 Hz	2500°	0.1%	
Temp Sensor 5		5.0	X	Anal	8°/sec	4 Hz	2500°	0.1%	
Temp Sensor 6		5.0	X	Anal	8°/sec	4 Hz	2500°	0.1%	
Temp Sensor 7		5.0	X	Anal	8°/sec	4 Hz	2500°	0.1%	
Rate Gyro Power	3.0		X	Disc.	dc	—	—	—	28V
Pitch Rate		5.0	X	Dig.	30 Hz	50 Hz	±30°/sec	0.1%	
Yaw Rate		5.0	X	Dig.	30 Hz	50 Hz	±30°/sec	0.1%	
Roll Rate		5.0	X	Dig.	60 Hz	50 Hz	±60°/sec	0.1%	
Rate Gyro Pwr	3.0		X	Disc.	400 Hz	—	—	—	26V rms
Pri Eng O/F Ratio	5.0	5.0	X	Dig.		50		30 Bits	
Pri Eng O/F Ratio	5.0	5.0	X	Anal	50	50		1%	
Preburn Valve Cmd (1)		2.1, 8.0	X	Anal	>30 Hz	>30 Hz		1%	
Preburn Valve Cmd (2)		2.1, 8.0	X	Anal	>30 Hz	>30 Hz		1%	
Preburn Valve Cmd (3)		2.1, 8.0	X	Anal	>30 Hz	>30 Hz		1%	
Preburn Valve Cmd (4)		2.1, 8.0	X	Anal	>30 Hz	>30 Hz		1%	
Preburn Valve Cmd (5)		2.1, 8.0	X	Anal	>30 Hz	>30 Hz		1%	
Preburn Valve Pos (1)	2.1	8.0	X	Anal	400	3 Hz		0.1%	
Preburn Valve Pos (2)	2.1	8.0	X	Anal	400	3 Hz		0.1%	
Preburn Valve Pos (3)	2.1	8.0	X	Anal	400	3 Hz		0.1%	
Preburn Valve Pos (4)	2.1	8.0	X	Anal	400	3 Hz		0.1%	

Note: Primary engine throttle signals reassigned to LH_2 flow control.

Data Points	Input From	Output To	Test Point	Characteristics					
				Type	Freq	Freq (Filtered)	Range	Resolution	Remarks
Preburn Valve Pos (5)	2.1	8.0	X %	Anal	400	3 Hz		0.1%	
Main Valve Cmd (1)		2.1, 8.0	X	Anal	>30 Hz	>30 Hz		1%	
Main Valve Cmd (2)		2.1, 8.0	X	Anal	>30 Hz	>30 Hz		1%	
Main Valve Cmd (3)		2.1, 8.0	X	Anal	>30 Hz	>30 Hz		1%	
Main Valve Cmd (4)		2.1, 8.0	X	Anal	>30 Hz	>30 Hz		1%	
Main Valve Cmd (5)		2.1, 8.0	X	Anal	>30 Hz	>30 Hz		1%	
Main Valve Pos (1)	2.1	8.0	X	Anal	400	3 Hz		0.1%	
Main Valve Pos (2)	2.1	8.0	X	Anal	400	3 Hz		0.1%	
Main Valve Pos (3)	2.1	8.0	X	Anal	400	3 Hz		0.1%	
Main Valve Pos (4)	2.1	8.0	X	Anal	400	3 Hz		0.1%	
Main Valve Pos (5)	2.1	8.0	X	Anal	400	3 Hz		0.1%	
Oxidizer Press. Valve Cmd (1)		2.1, 8.0	X	Anal	>30 Hz	>30 Hz		1%	
Oxidizer Press. Valve Cmd (2)		2.1, 8.0	X	Anal	>30 Hz	>30 Hz		1%	
Oxidizer Press. Valve Cmd (3)		2.1, 8.0	X	Anal	>30 Hz	>30 Hz		1%	
Oxidizer Press. Valve Cmd (4)		2.1, 8.0	X	Anal	>30 Hz	>30 Hz		1%	
Oxidizer Press. Valve Cmd (5)		2.1, 8.0	X	Anal	>30 Hz	>30 Hz		1%	
Oxidizer Press. (1)	2.1	8.0	X	Anal	30 Hz	3 Hz		0.1%	
Oxidizer Press. (2)	2.1	8.0	X	Anal	30 Hz	3 Hz		0.1%	
Oxidizer Press. (3)	2.1	8.0	X	Anal	30 Hz	3 Hz		0.1%	
Oxidizer Press. (4)	2.1	8.0	X	Anal	30 Hz	3 Hz		0.1%	
Oxidizer Press. (5)	2.1	8.0	X	Anal	30 Hz	3 Hz		0.1%	

6.0 Communication

Data Points	Input From	Output To	Test Point	Characteristics				
				Type	Freq	Freq (Filtered)	Range	Resolution
Mike Audio	7.0		X	Anal	30-3K	—	0-0.5V	0.002V
E/P Audio		7.0	X	Anal	30-3K		0-0.5V	0.002V
Vox Sens Cont	7.0		X	Anal	dc		0.28V	0.2V
I/C-PTT Cont	7.0		X	On-Off	dc		0-28V	10%
Vox Cont	7.0		X	On-Off	dc		0-28V	10%
Vox/PTT Cont	7.0		X	On-Off	dc		0-28V	10%
PTTSW Cont	7.0		X	On-Off	dc		0-28V	10%
E/R Vol Cont	7.0		X	Anal	dc		0-28V	0.2V
S/K Vol Cont	7.0		X	Anal	dc		0-28V	0.2V
S/K Trans Cont	7.0		X	On-Off	dc		0-28V	10%
S/K Rer Cont	7.0		X	On-Off	dc		0-28V	10%
UHF Vol Cont	7.0		X	Anal	dc		0-28V	0.2V
UHF Xmit Cont	7.0		X	On-Off	dc		0-28V	10%

Data Points	Input From	Output To	Test Point	Characteristics				
				Type	Freq	Freq (Filtered)	Range	Resolution
UHF Rer Cont	7.0		X	On-Off	dc		0-28V	10%
I/C Vol Cont	7.0		X	Anal	dc		0-28V	0.2V
I/C Xmit Cont	7.0		X	On-Off	dc		0-28V	10%
I/C Rer Cont	7.0		X	On-Off	dc		0-28V	10%
Pwr Cont	7.0		X	On-Off	dc		0-28V	10%
Data from DM	8.0		X	Dig. Word	3kbs		0-5V	10%
Data to DM		8.0	X	Dig. Word	3kbs		0-5V	10%
L/F Clock DM	8.0		X	Anal	400		0-5V	10%
Clock DM	8.0		X	Dig. Word	12kbs		0-5V	10%
Pwr	3.0			dc	—		0-28V	10%

7.0 Control and Displays

Data Points	Characteristics				
	Type	Freq (TP)	Freq (Filtered)	Range	Resolution
Power Supply Voltage	Voltage	dc	—	5V	1%
Power Supply Voltage	Voltage	dc		28V	1%
Power Supply Voltage	Voltage	dc		15V	1%
Power Supply Voltage	Voltage	dc		-15V	1%
Power Supply Voltage	Voltage	dc		10kV	2%
Power Supply Voltage	Voltage	dc		10kV	2%
Power Supply Voltage	Voltage	dc		10kV	2%
Memory Waveform	Waveform	1MHz			
Sweep Waveform	Waveform	Variable		0-1MHz	
Digitizer "Zero"	Digital			7 bits	
Digitizer "Full Scale"	Digital			7 bits	
CRT Display Voltages	Voltage			0-10kV	2%
CRT Display Voltages	Voltage			0-10kV	2%
CRT Display Voltages	Voltage			0-10kV	2%
CRT Display Voltages	Voltage			0-10kV	2%
CRT Display Voltages	Voltage			0-10kV	2%
CRT Display Voltages	Voltage			0-10kV	2%
Logic Pulsetrains	Waveform	Variable		0-1MHz	
Logic Pulsetrains	Waveform	Variable		0-1MHz	
Logic Pulsetrains	Waveform	Variable		0-1MHz	
Logic Pulsetrains	Waveform	Variable		0-1MHz	
Logic Pulsetrains	Waveform	Variable		0-1MHz	

APPENDIX B. SHUTTLE VEHICLE CHARACTERISTICS

To define the boundaries and constraints on the avionics system, contractor vehicle configurations were selected as representative conceptual designs. These configurations were used as necessary to provide input data for the avionics system design. The control systems studies are the chief users of such data; therefore, the format is generally tailored to providing configuration information in a form most suited to stability and control analysis requirements.

Tables B-1 through B-3 give the mass, moment of inertia, center of gravity, and other important geometric parameters of the vehicle for initial and final flight conditions for each stage and for the combined booster and orbiter. Position reference axes are noted on the configuration drawings shown in Section II.

Figures B-1 through B-32 show the available aerodynamic data. In general, the static derivatives are contractor data, or are derived from contractor data at a few Mach numbers.

It is emphasized that the available aerodynamic data shown is almost entirely based on theory and is subject to significant change as wind tunnel test data are obtained. Table B-4 shows the available data for each vehicle and stage. The number in the matrix is the figure number for the appropriate vehicle and derivative.

TABLE B-1. LOCKHEED CONFIGURATION MASS AND GEOMETRIC CHARACTERISTICS

	Combination		Booster			Orbiter			
	Launch	Staging	Staging	Entry	Landing	Staging	On Orbit	Entry	LNDG
Weight (lb)	3 734 906		412 340		374 837	978 269	308 762		259 500
C. G. (ft)									
X	96.8		162.9		159.8	142.6	154.0		151.9
Z	3.6		-0.5		-0.6	26.2	28.4		29.5
I_x (slug ft^2 × 10^{-6})	36.54		6.15		6.15	8.13	2.37		2.27
I_y	377.55		57.3		54.92	24.7	15.14		14.1
I_z	363.2		57.84		55.45	31.78	16.75		15.75
I_{xz}	.54250				-.020	-.025	.102		-.045
I_{yz}									
I_{xy}									
L_{REF} (ft)	206		206			164			
S_{REF} (ft^2)	12 000		12 000			5800			
Main Engine Thrust S. L. (lb)			534 500 (11)			518 000 (2)			
VAC			581 800			590 000			
I_{SP} S. L.			392			388			
VAC			428.5			454.2			
TVC Angle (deg)									
Gimbal Point ft. X						161.3			
δ Max. Elevon (deg)						25			
δ Max. Rudder (deg)									
Air Breathing Engine Thrust SLTO (lb)			63 600 (4)			46 000			

TABLE B-2. MASS AND GEOMETRIC CHARACTERISTICS, MARTIN CONFIGURATION

	Combination		Booster			Orbiter			
	Launch	Staging	Staging	Entry	Landing	Staging	On Orbit	Entry	LNDG
Weight (lb)	5 390 553	1 113 440	497 213			616 225	231 900		
CG (ft)									
X		119	126			115.1	124		
Z		12	19			7.5	10.3		
I_x (slug ft^2 × 10^{-6})		16.886	13.329			2.682	2.280		
I_y		53.090	34.470			16.475	12.080		
I_z		63.900	45.274			17.608	13.360		
I_{xz}		0.007513	0.003173				0.000087		
I_{yz}									
I_{xy}									
L_{REF} (ft)	191		191			170			
S_{REF} (ft^2)	11 200		11 200			7085			
Main Engine Thrust S. L. (lb)			500 000 (12)			500 000 (2)			
VAC			568 133			568 133			
I_{sp} S. L.			401						
VAC			456.5			456.5			
TVC Angle (deg)			□7			□7			
Gimbal Point (ft), X			183			158			
δ_{Max} Elevon (deg)									
δ_{Max} Rudder (deg)									
Air Breathing Engine Thrust SLTO (lb)			10 000 (8)			10 000 (4)			

TABLE B-3. MASS AND GEOMETRIC CHARACTERISTICS, GD/C CONFIGURATION — FR-3

	Combination		Booster			Orbiter			
	Launch	Staging	Staging	Entry	Landing	Staging	On Orbit	Entry	Landing
Weight (lb)	4 329 000	1 517 121	590 437	564 247	517 331	926 684	334 444	287 873	285 005
CG (ft)									
X	77.8		117.3	114.9	123.6	73.6	95.6	102.7	102.7
Z	-7.5		-2.6	-2.8	-2.9	-1.7	-1.7	-0.9	-1.4
I_x (slug ft^2 x 10^{-6})	36.39		4.75	4.74/9.84	9.78	2.78	2.24	2.04/3.05	3.12
I_y	356.69		98.64	96.02/92.90	74.66	51.84	27.80	23.55/22.66	22.29
I_z	357.39		99.14	96.51/98.51	80.04	52.15	28.13	23.80	23.38
I_{xz}	1.093		2.94	2.82/3.58	2.8	0.394	0.823	0.479/0.670	0.626
I_{xz}									
I_{xy}									
L_{REF} (ft)	210.1		210.1			179.2			
S_{REF} (ft^2)	6100		6100						
Main Engine Thrust S. L.(lb)			400 000 (15)			400 000 (3)			
VAC			462 000			472 000			
I_{sp} S. L.			389.3			232			
VAC			449.5			459			
TVC Angle (deg)			5 deg						
Gimbal Point (ft)									
X			201.1 - 209.15						
δ_{Max} Elevon (deg)									
δ_{Max} Rudder (deg)			± 10						
Air Breathing Engine Thrust SLTO (lb)			52 500 (4)			21 000 (3)			

TABLE B-4. AERODYNAMIC DATA
M = 0 → 10 (LAUNCH AND BOOSTER) AND M = 0 - 20 (ORBITER)

Derivative	Lockheed			GD/C			Martin		
	Launch	Booster	Orbiter	Launch	Booster	Orbiter	Launch	Booster	Orbiter
Longitudinal									
$CN\alpha$	Fig. B-31	Fig. B-29							Fig. B-14
CNq									
$Cm\alpha$									
Cmq									
Cmo									
CDo	Fig. B-32	Fig. B-30							
$CD\alpha$									
CLu									
Cmu									
Cxu									
Lateral									
$Cy\beta$		Fig. B-28							Fig. B-13
$Cn\beta$		Fig. B-27							Fig. B-12
$C\ell\beta$		Fig. B-26							Fig. B-11
Cyp		Fig. B-25							Fig. B-10
Cnp		Fig. B-24							Fig. B-9
$C\ell p$		Fig. B-23							Fig. B-8
Cyr		Fig. B-22							Fig. B-7
Cnr		Fig. B-21							Fig. B-6
$C\ell r$		Fig. B-20							Fig. B-5
Control									
$CL_{\delta e}$		Fig. B-19							Fig. B-4
$Cm_{\delta e}$		Fig. B-18							Fig. B-3
$Ch_{\delta e}$		Fig. B-16							Fig. B-1
$Ch_{\alpha e}$									
$C\ell_{\delta e}$		Fig. B-17							Fig. B-2
$Cy_{\delta R}$		Fig. B-19							Fig. B-4
$C\ell_{\delta R}$		Fig. B-17							Fig. B-2
$Cn_{\delta R}$		Fig. B-18							Fig. B-3
$Ch_{\delta R}$		Fig. B-15							Fig. B-1
$Ch_{\beta R}$									

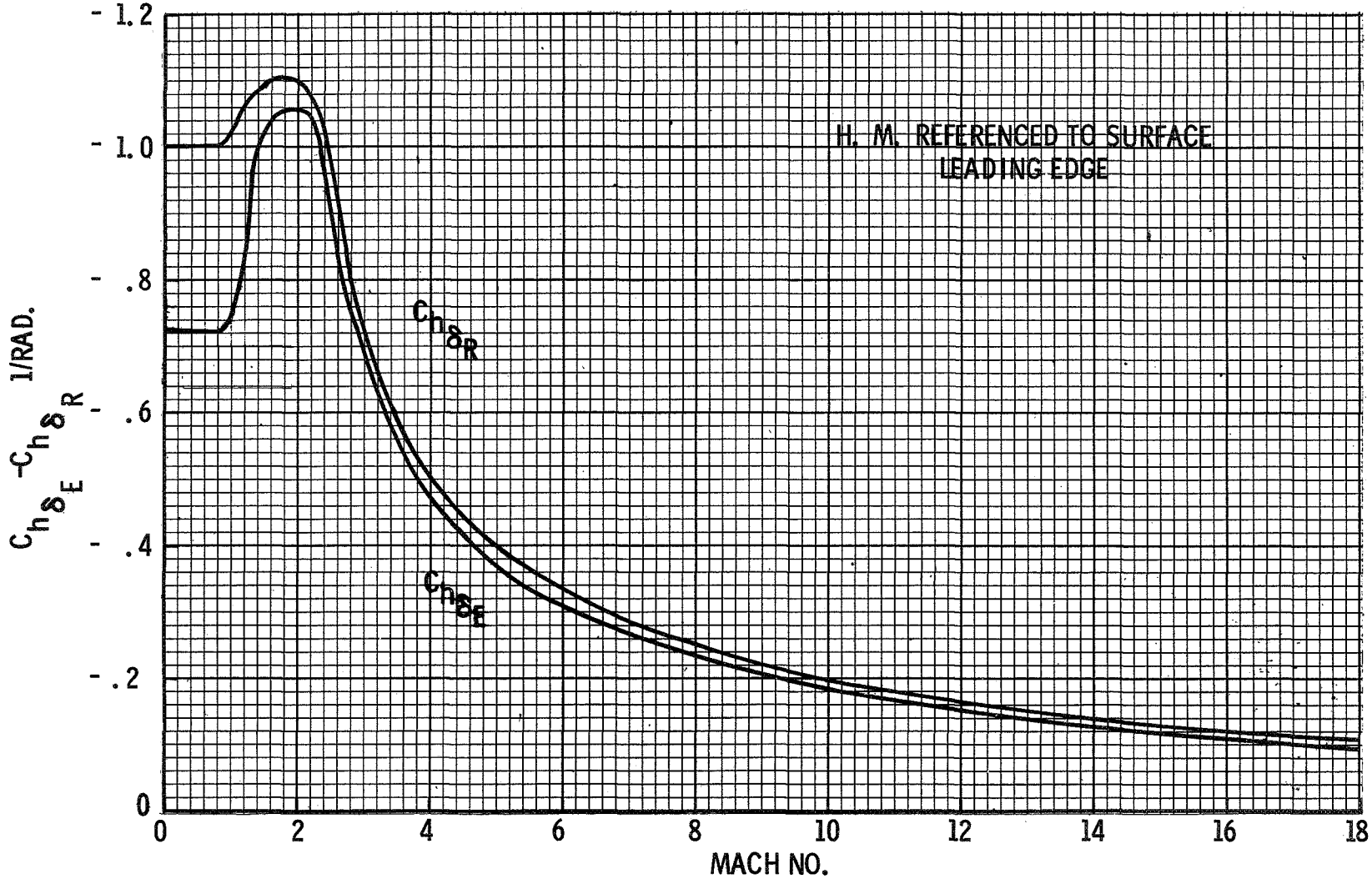


Figure B-1. Martin orbiter control surface hinge moment.

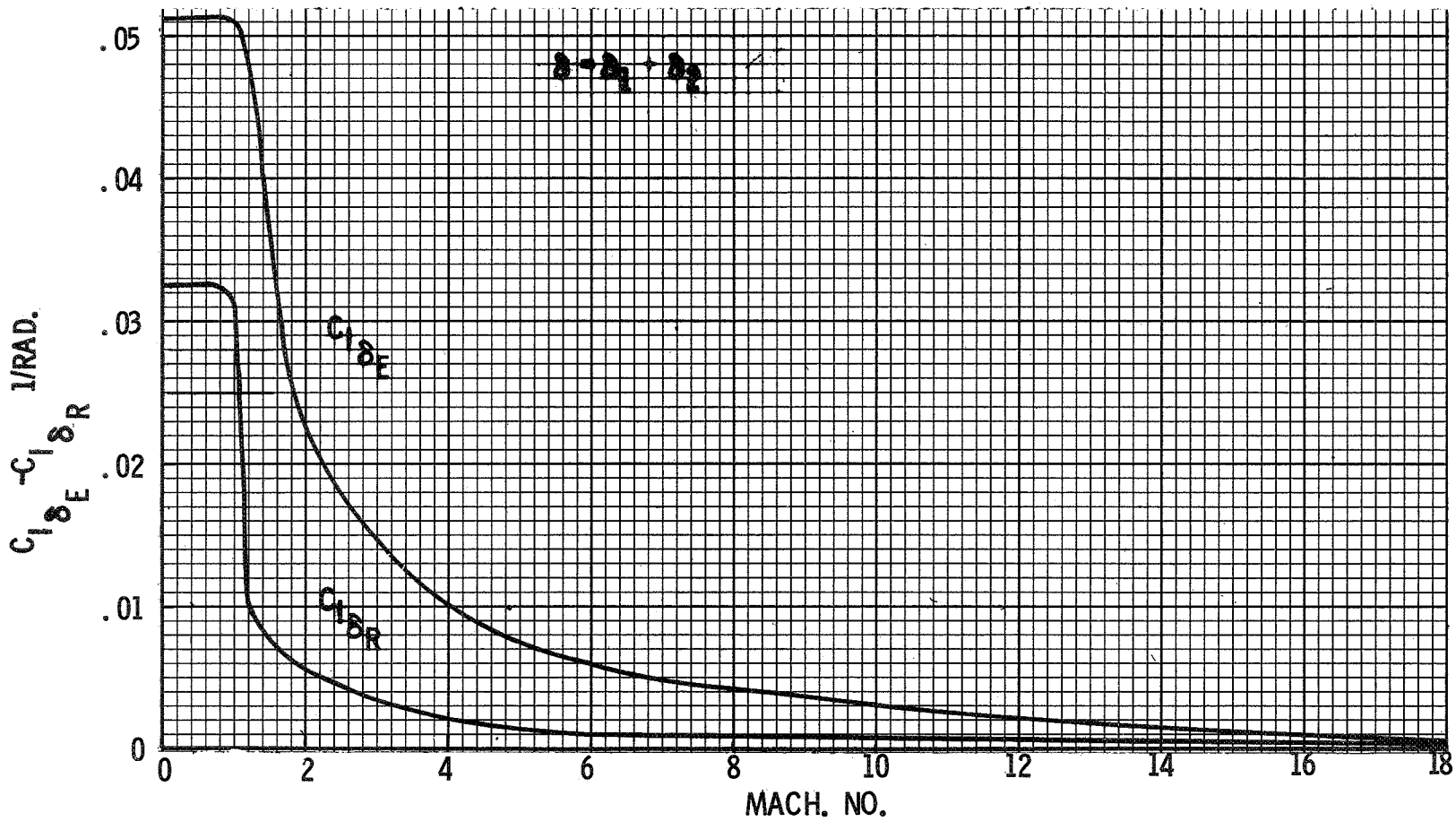


Figure B-2. Martin orbiter control surface effectiveness.

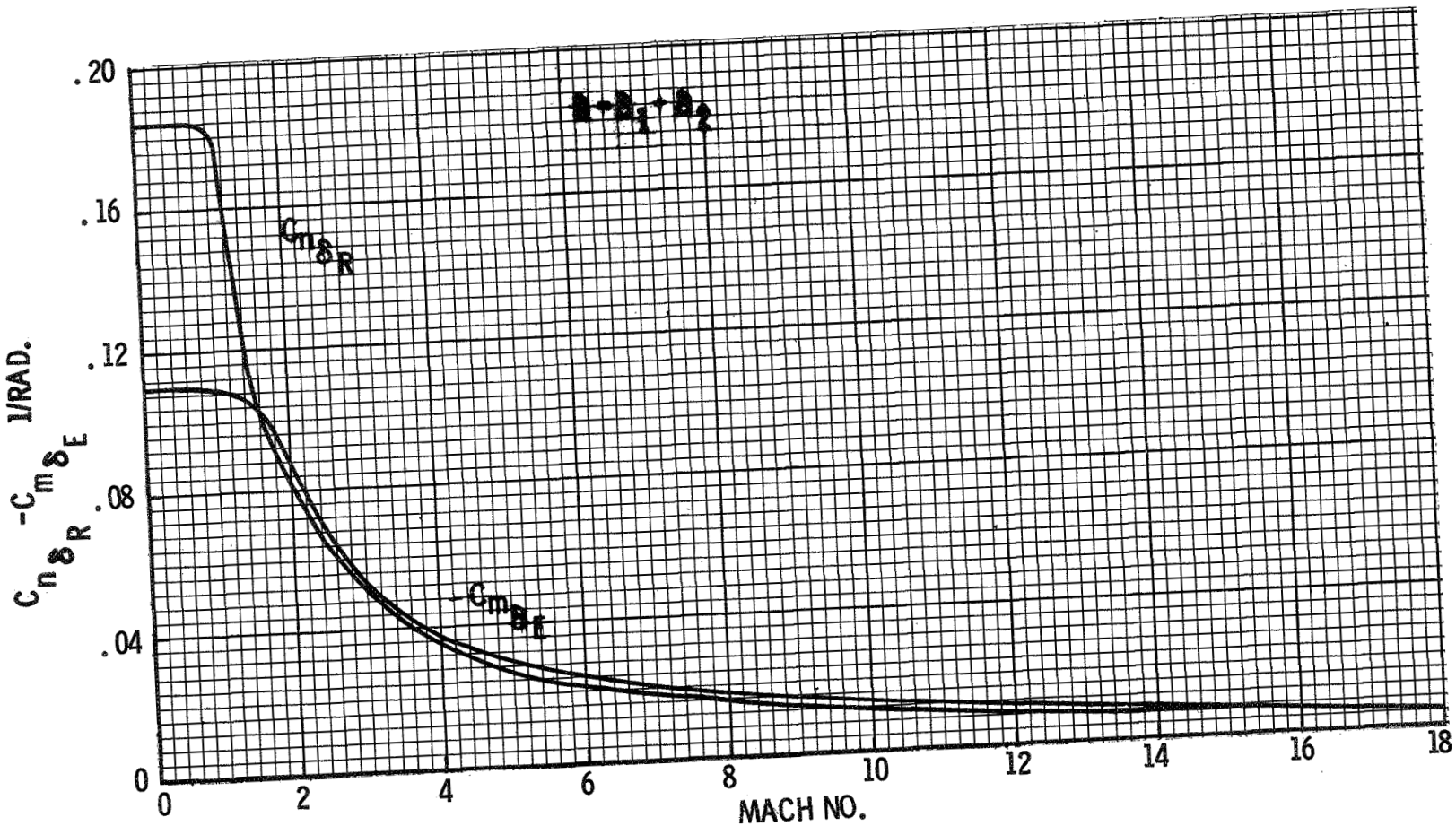


Figure B-3. Martin orbiter control surface effectiveness.

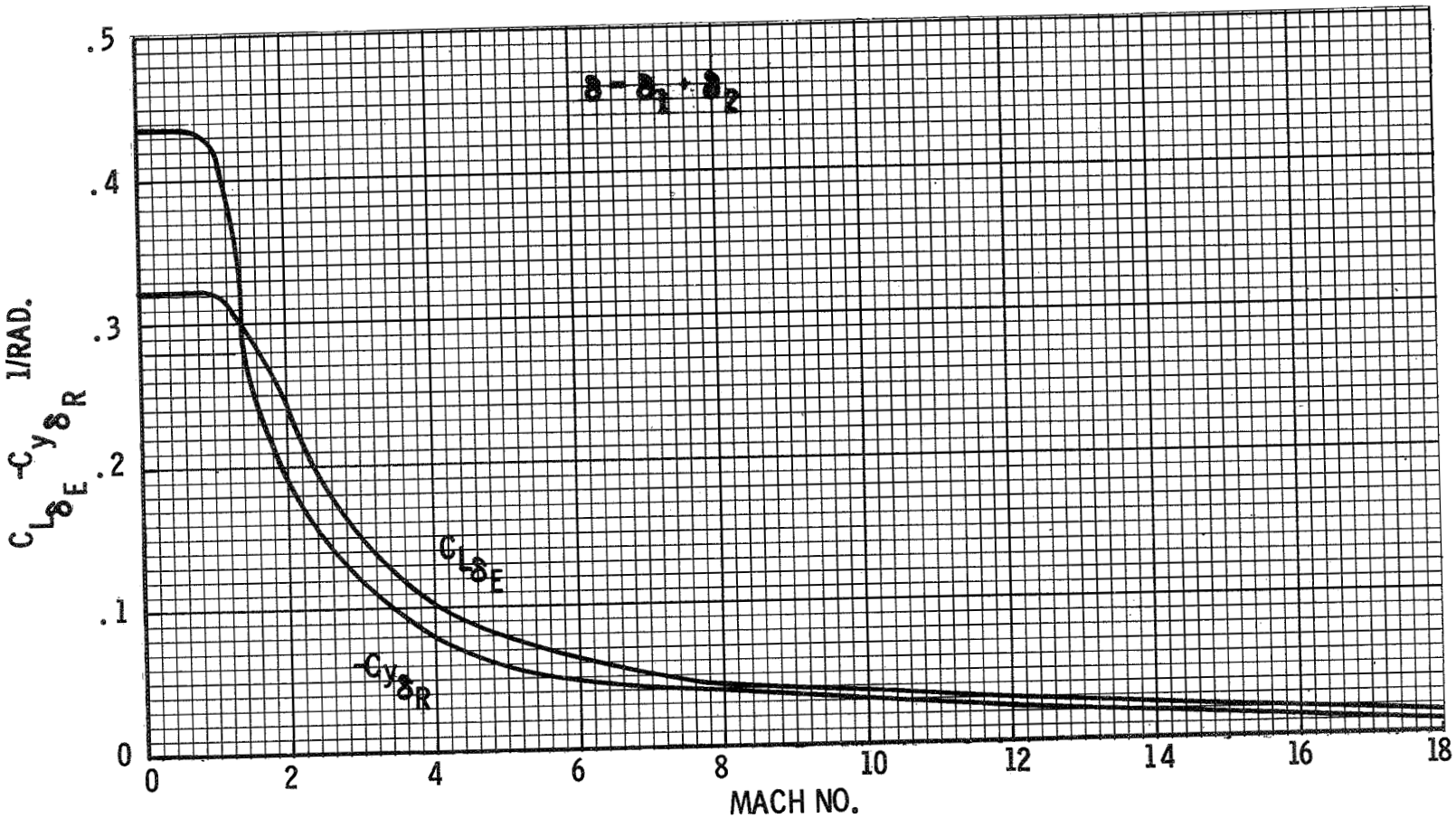


Figure B-4. Martin orbiter control surface effectiveness.

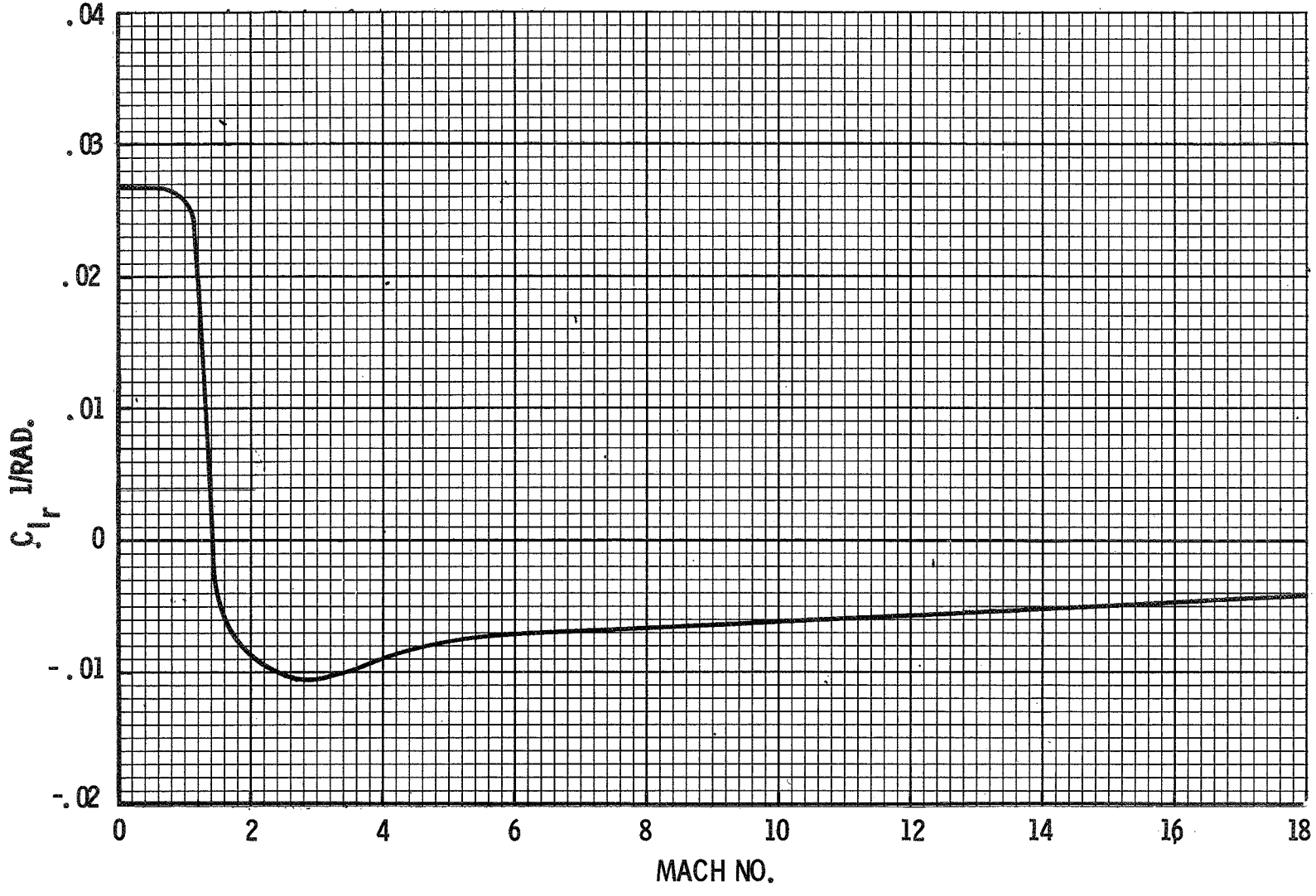


Figure B-5. Martin orbiter rolling moment due to yaw rate.

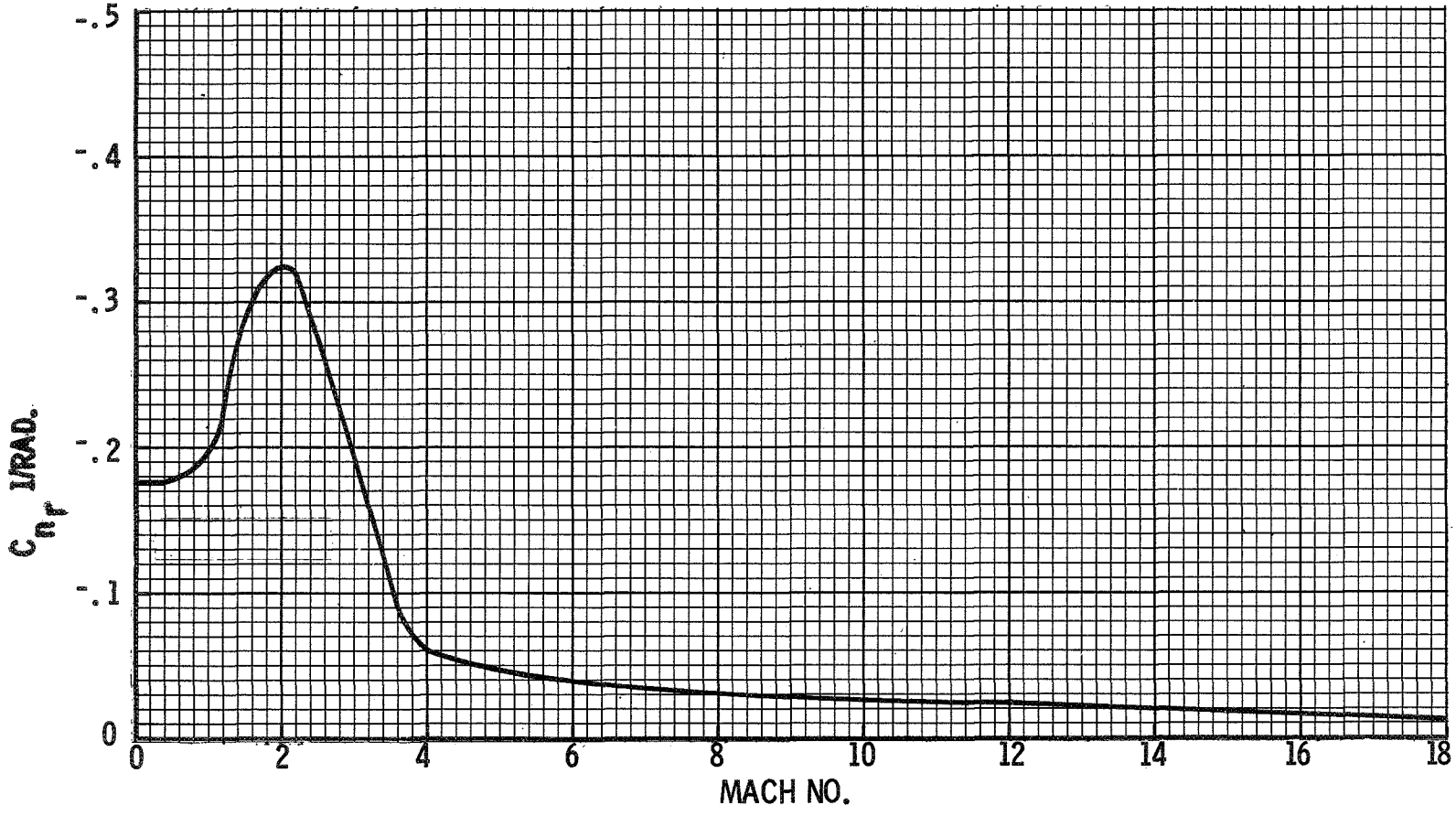


Figure B-6. Martin orbiter damping in yaw.

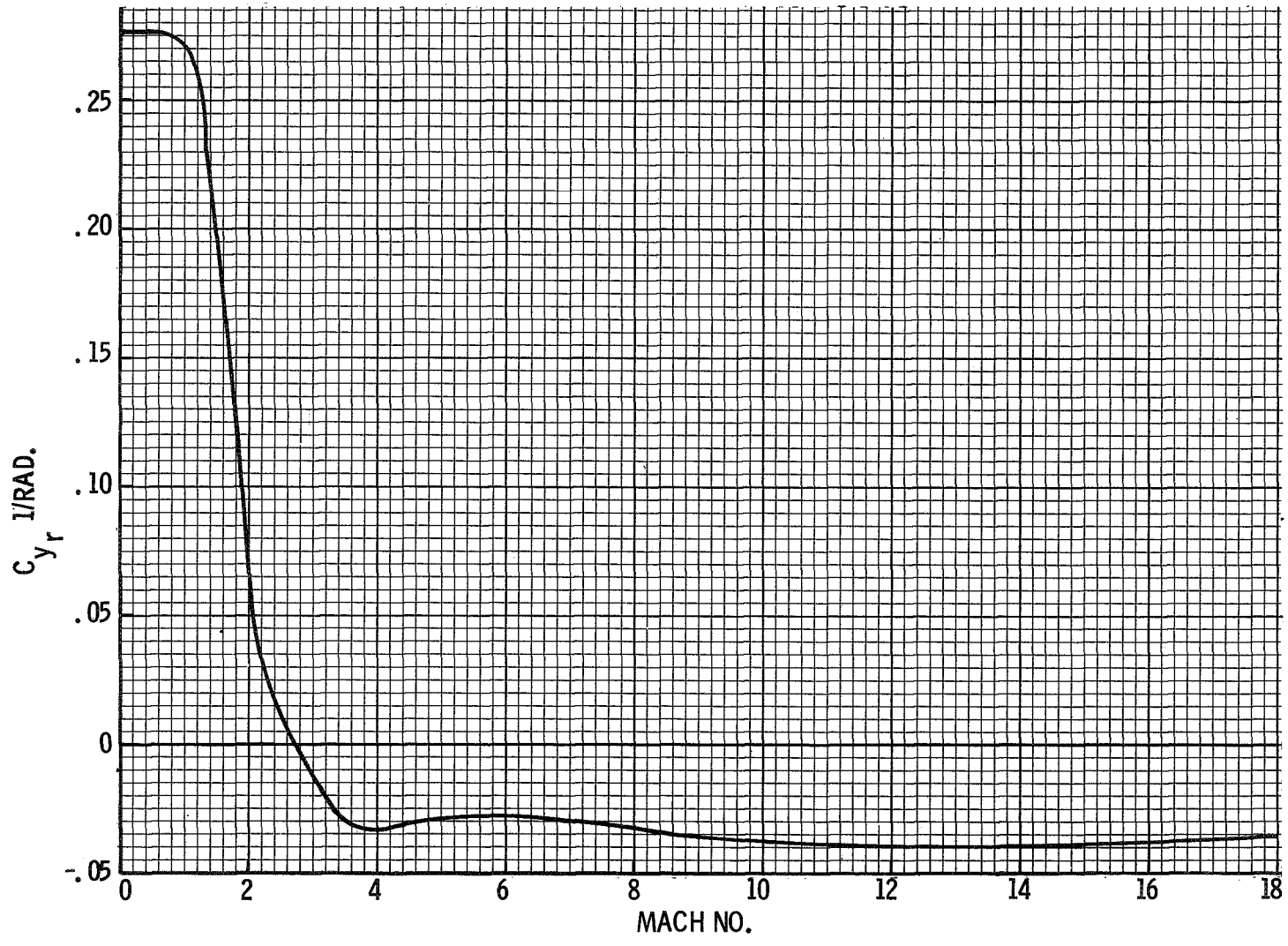


Figure B-7. Martin orbiter side force due to yaw rate.

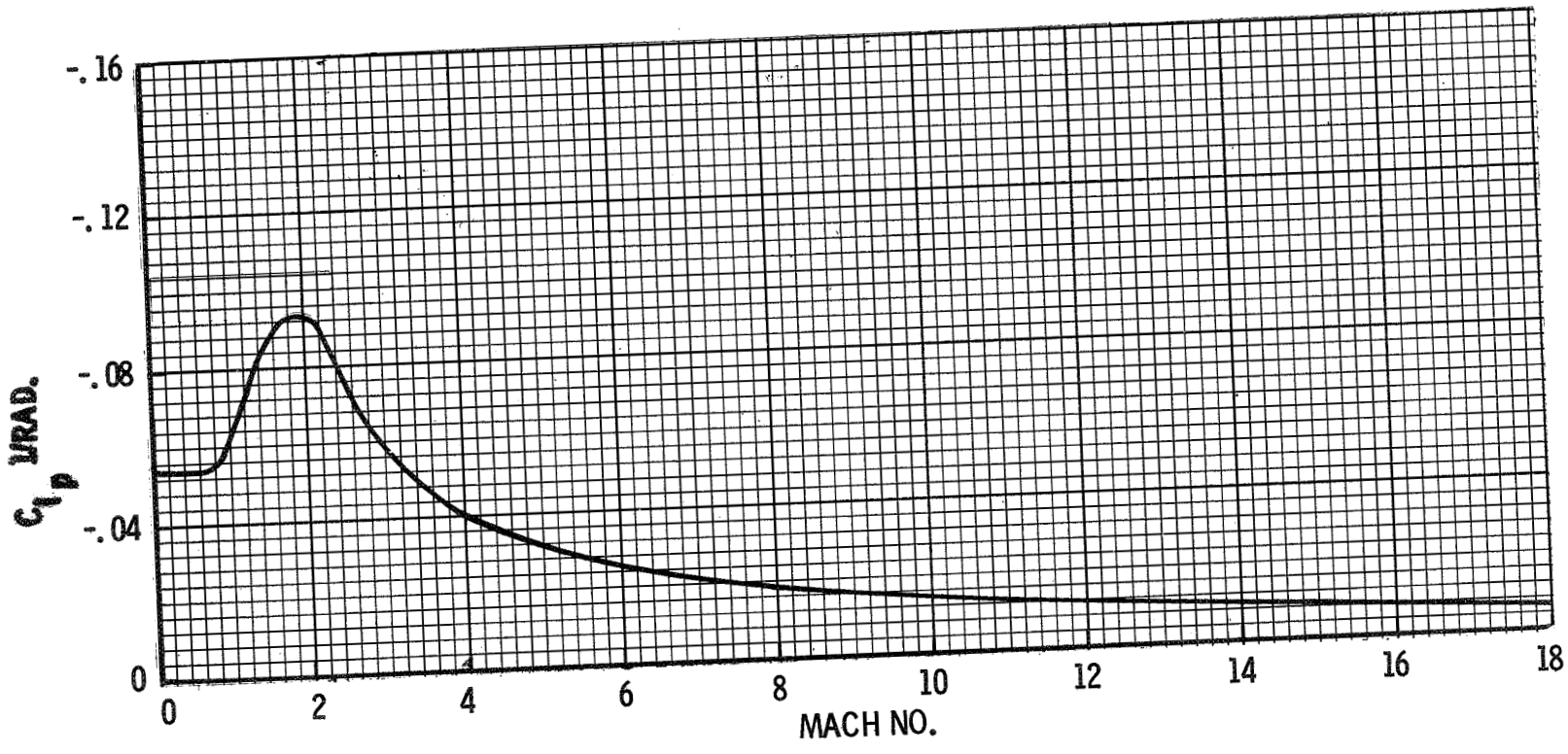


Figure B-8. Martin orbiter damping in roll.

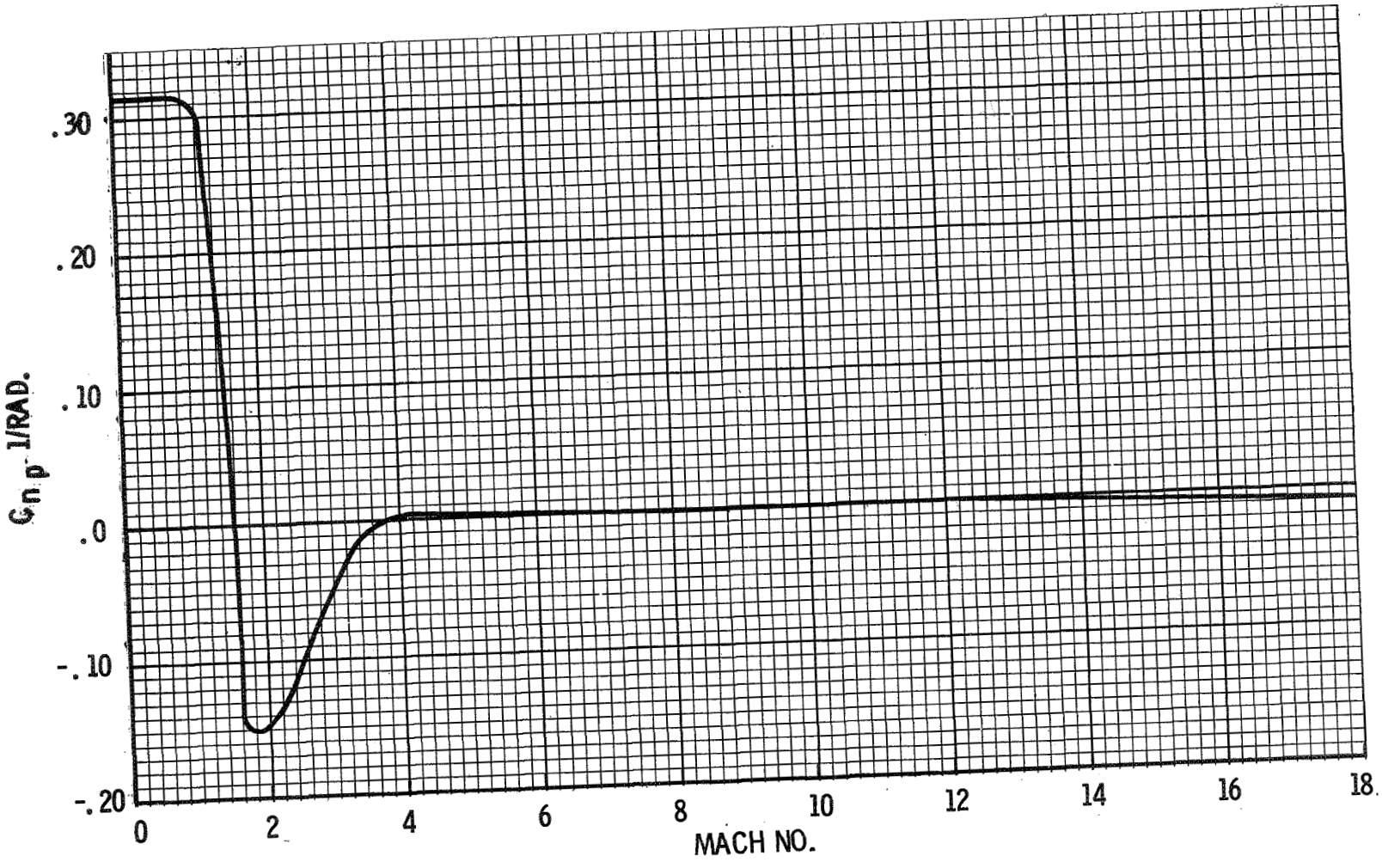


Figure B-9. Martin orbiter yawing moment due to roll rate.

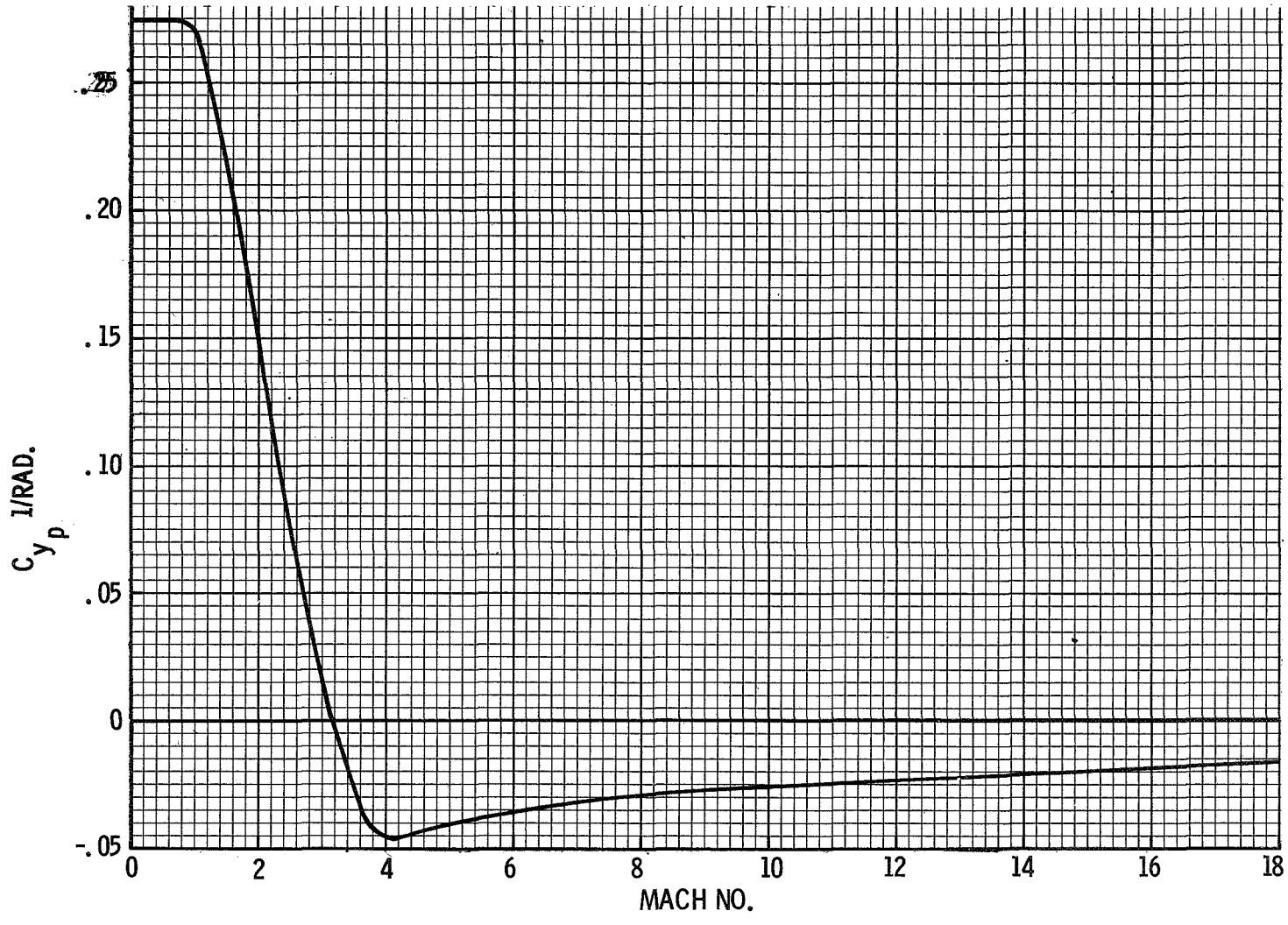


Figure B-10. Martin orbiter side force due to roll rate.

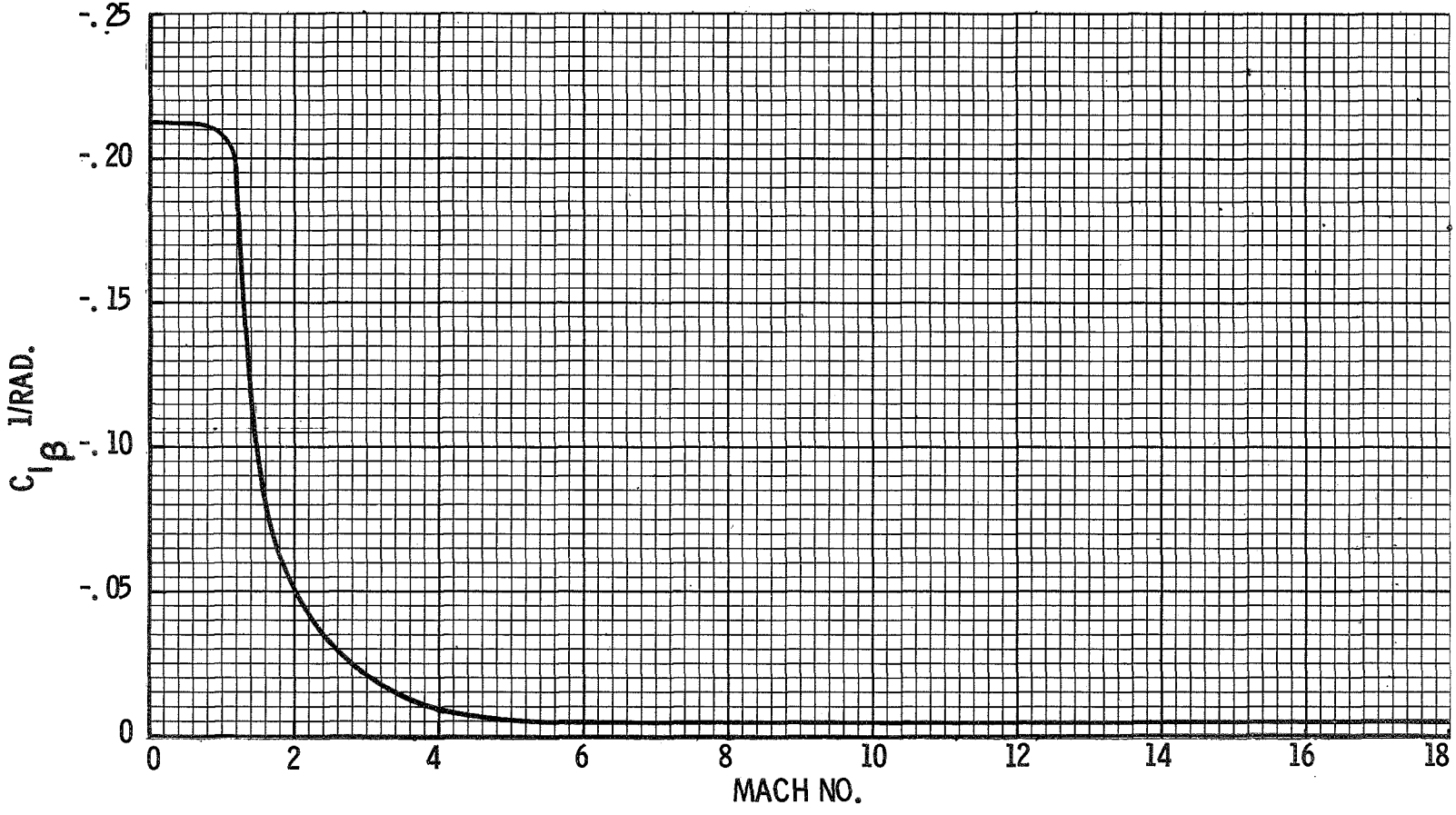


Figure B-11. Martin orbiter rolling moment due to sideslip.

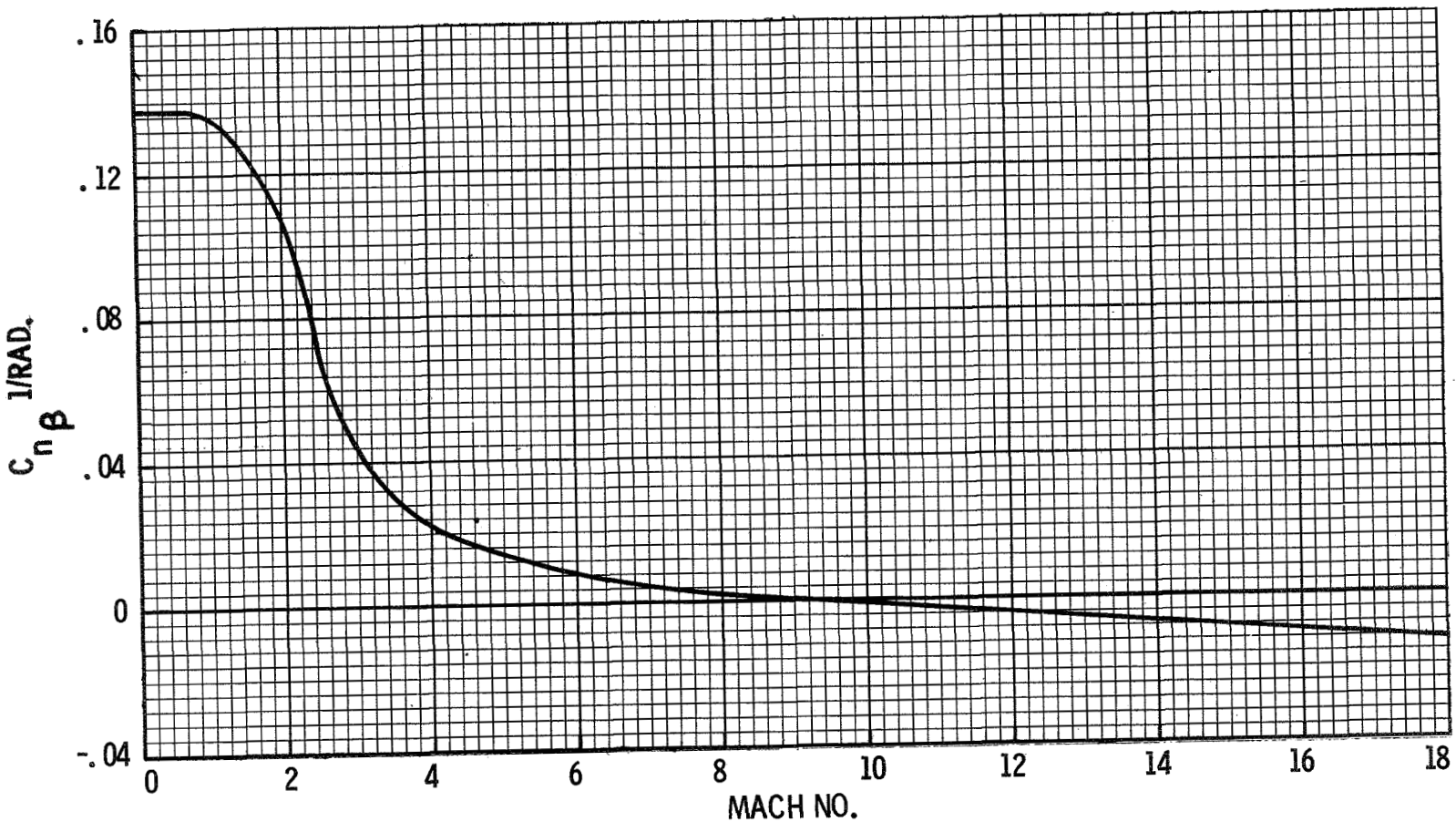


Figure B-12. Martin orbiter static directional stability.

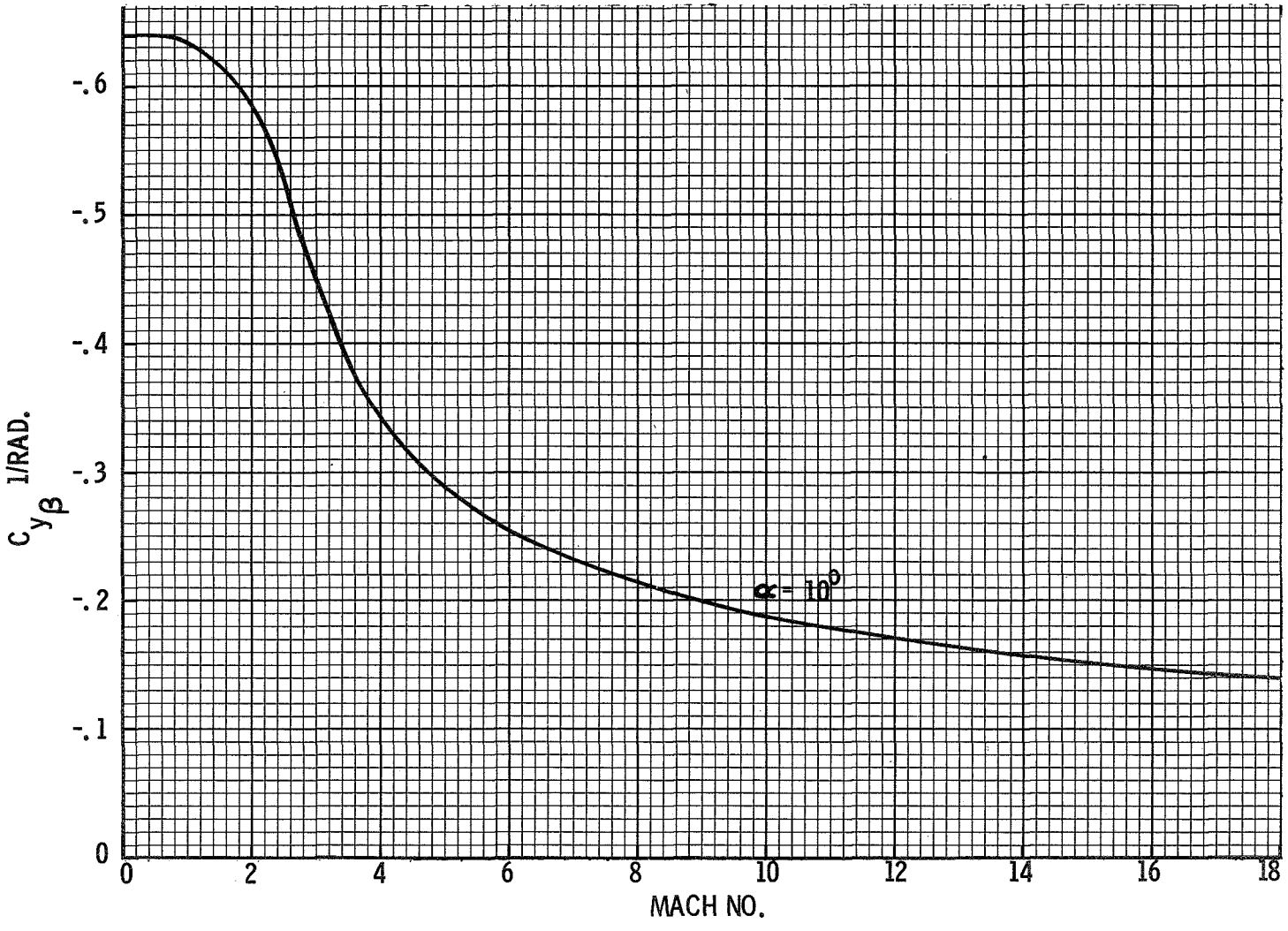


Figure B-13. Martin orbiter side force due to sideslip.

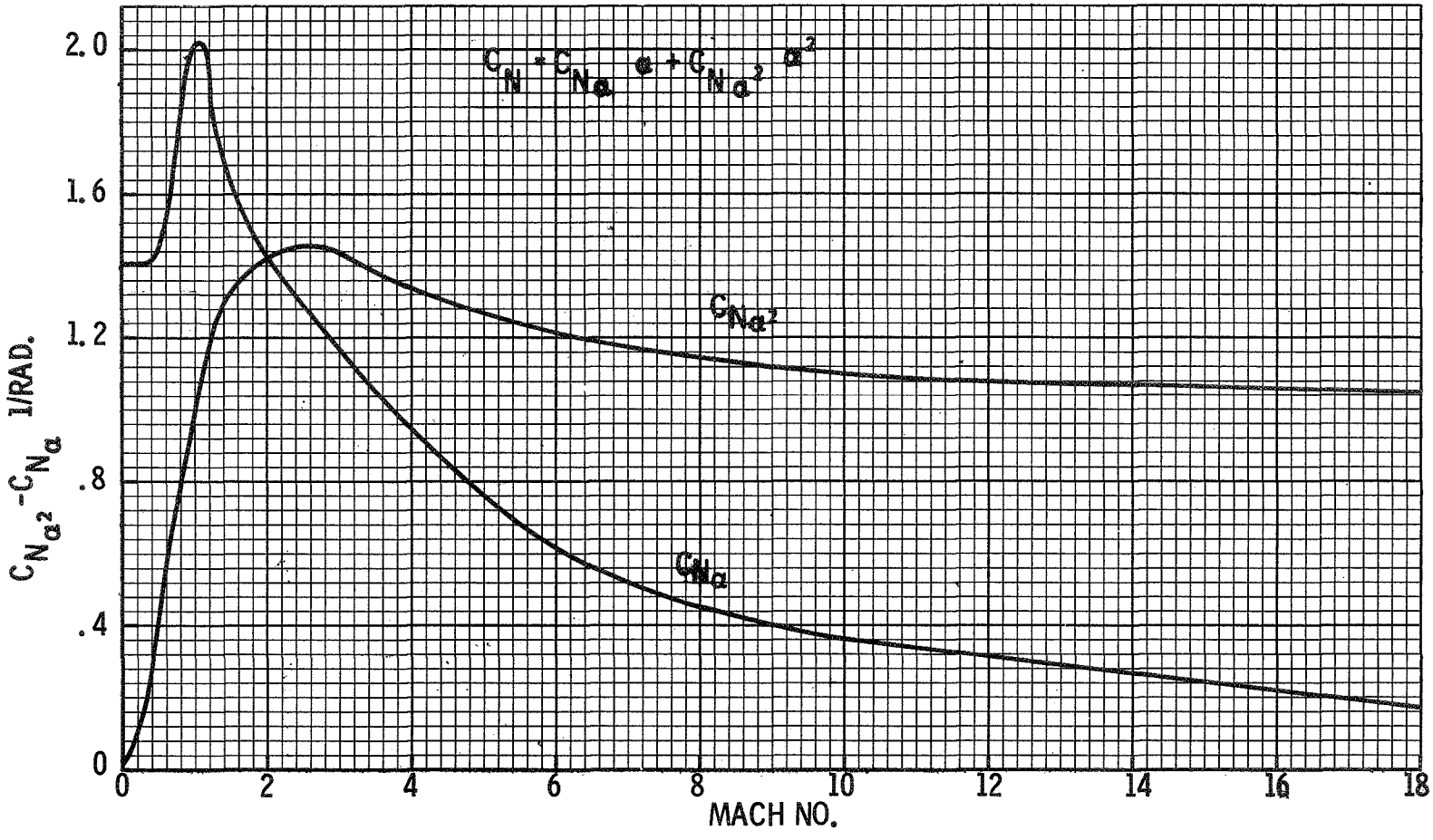


Figure B-14. Martin orbiter lift curve slope.

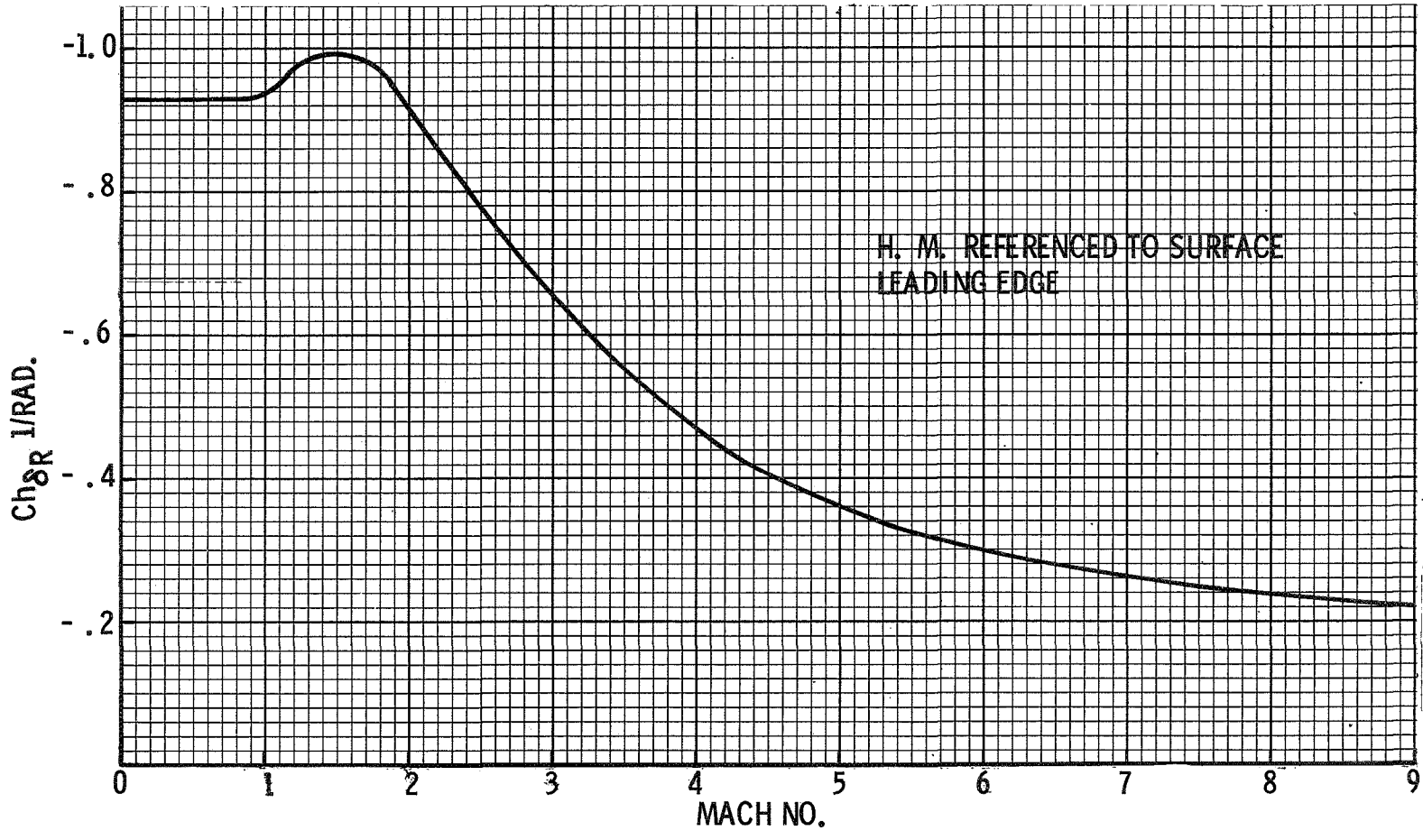


Figure B-15. Lockheed booster control surface hinge moment.

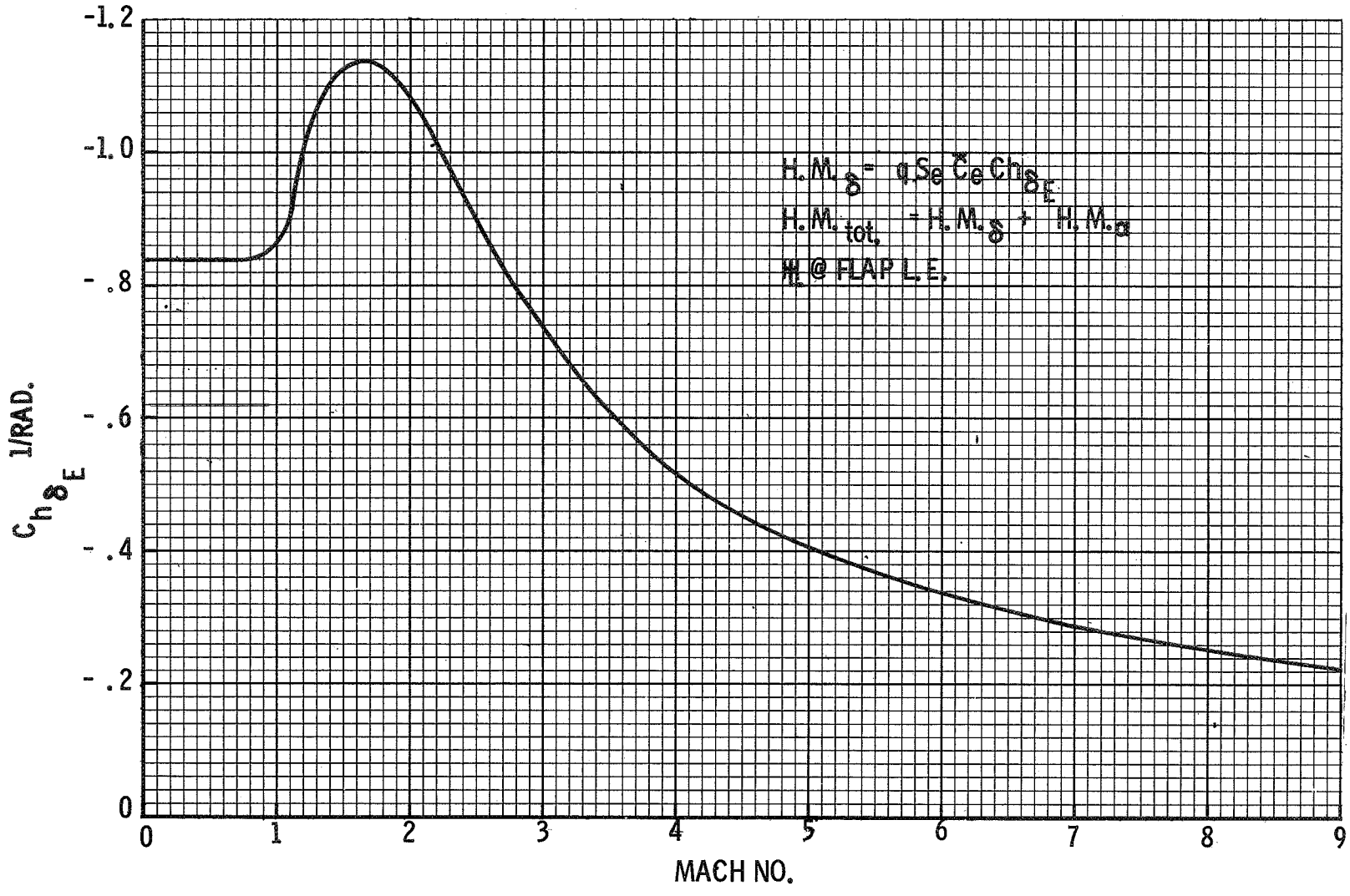


Figure B-16. Lockheed booster control surface hinge moment.

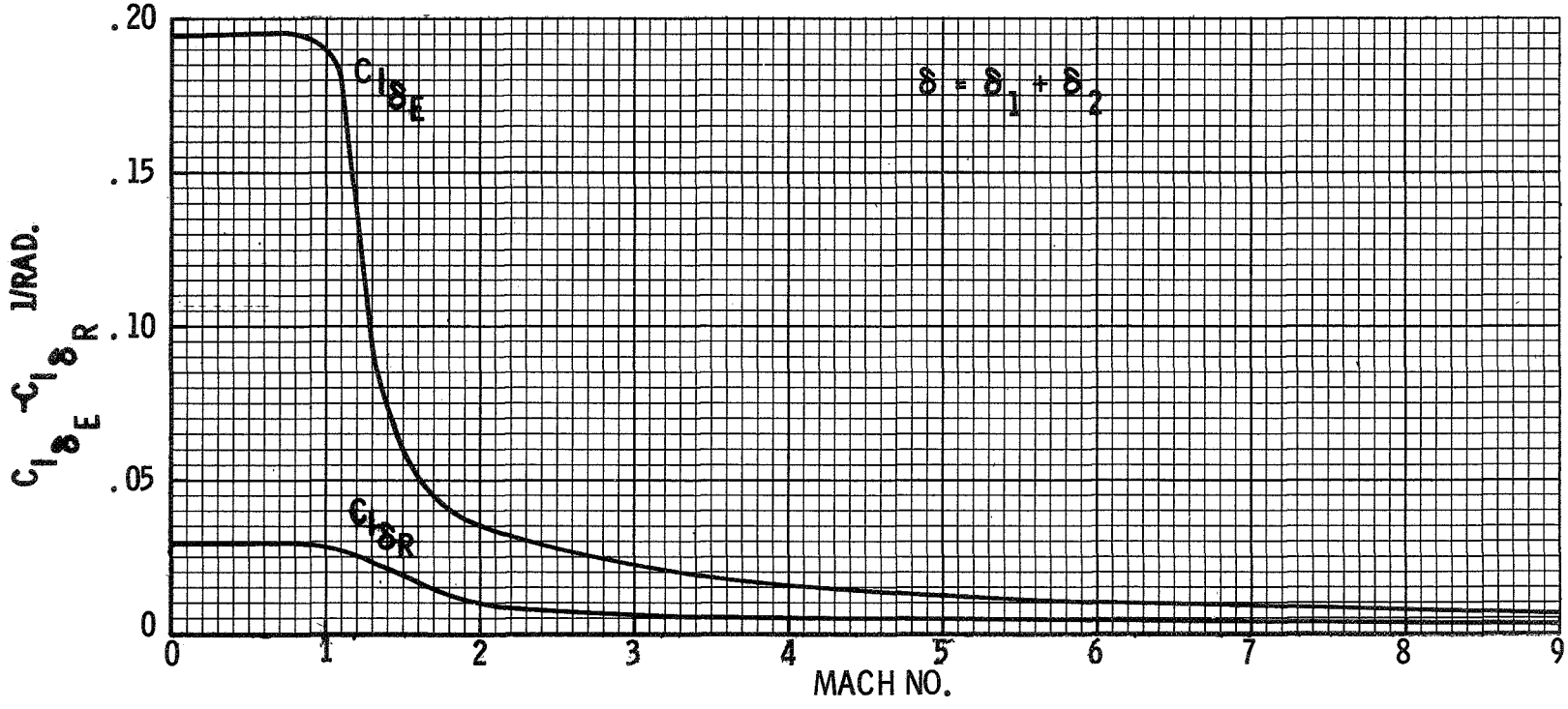


Figure B-17. Lockheed booster control surface effectiveness.

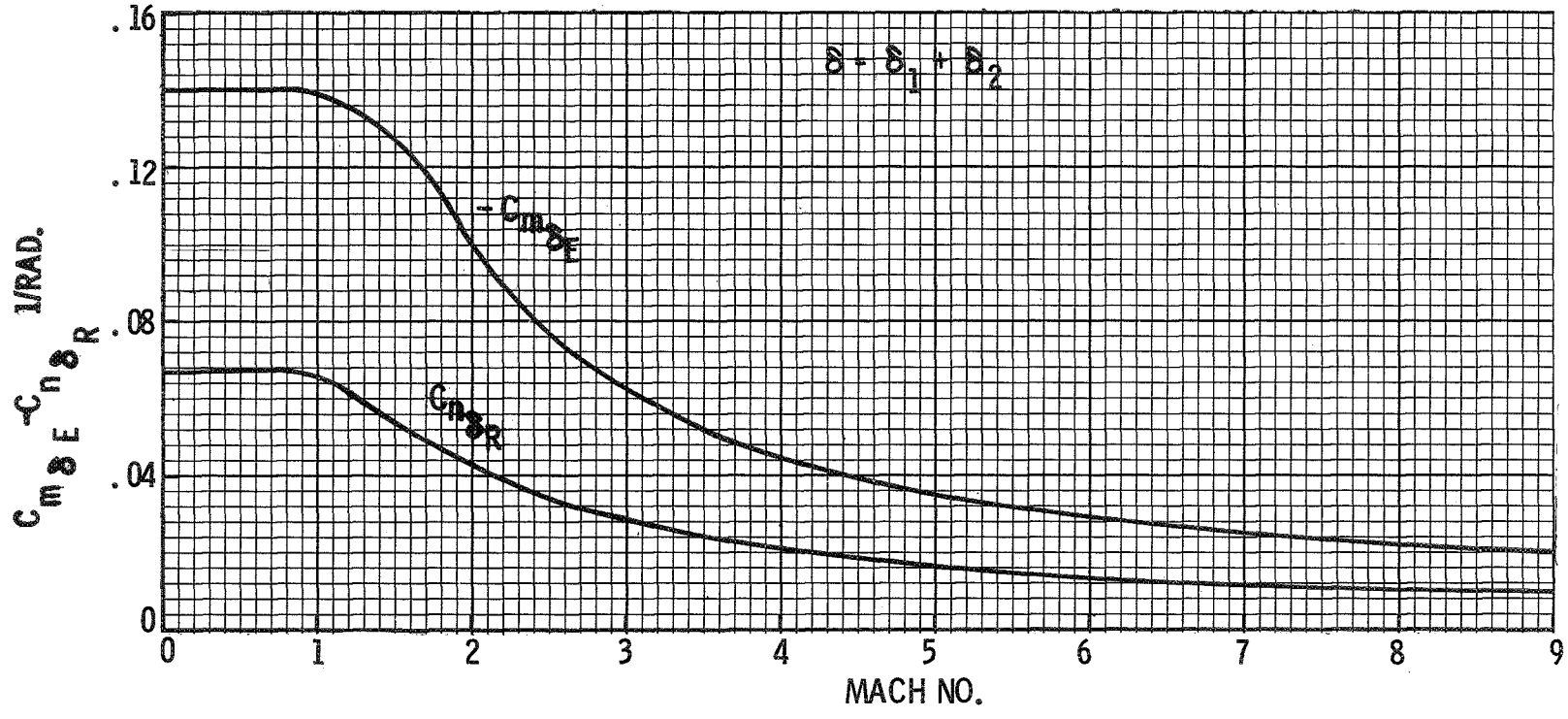


Figure B-18. Lockheed booster control surface effectiveness.

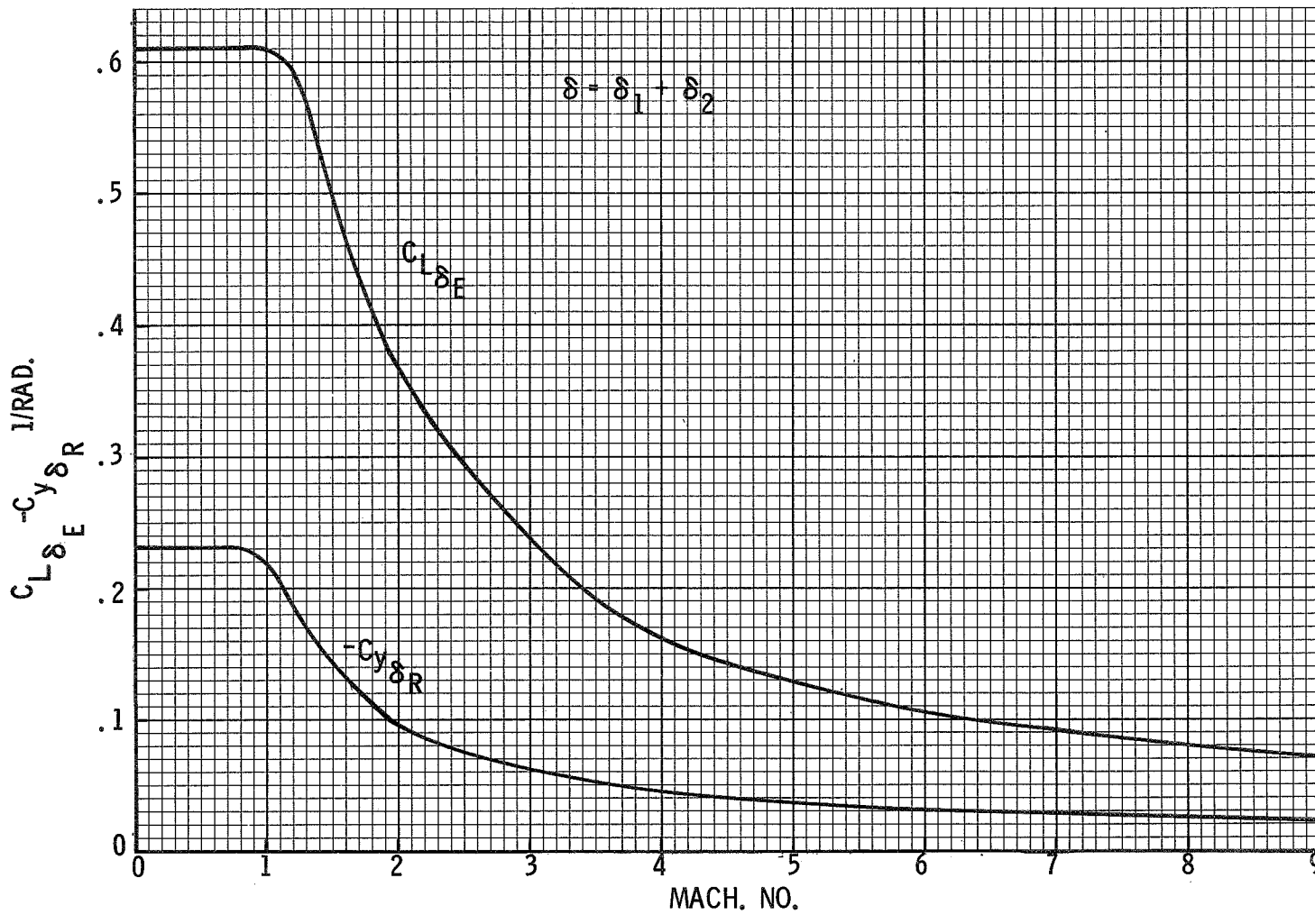


Figure B-19. Lockheed booster control surface effectiveness.

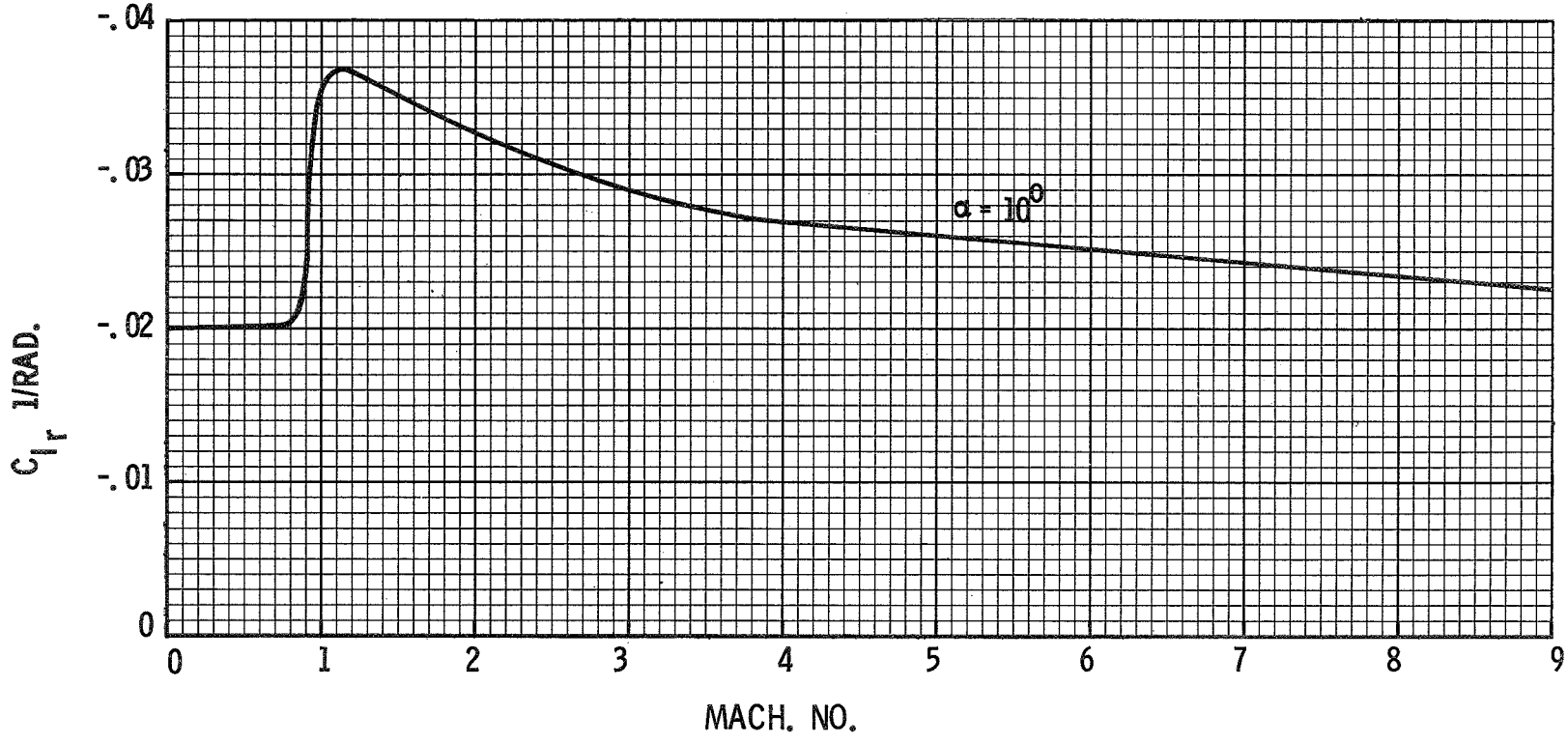


Figure B-20. Lockheed booster rolling moment due to yaw rate.

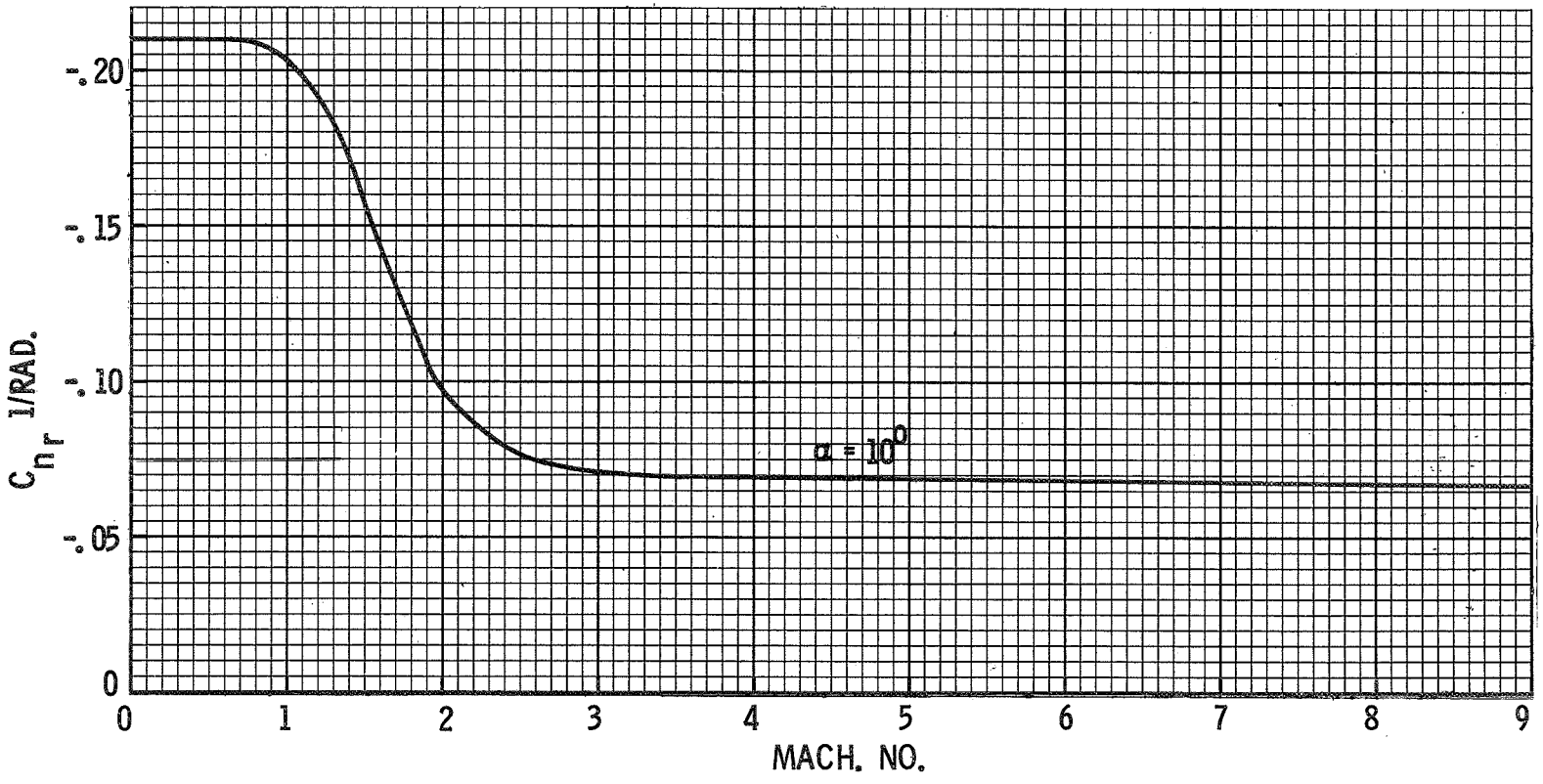


Figure B-21. Lockheed booster damping in yaw.

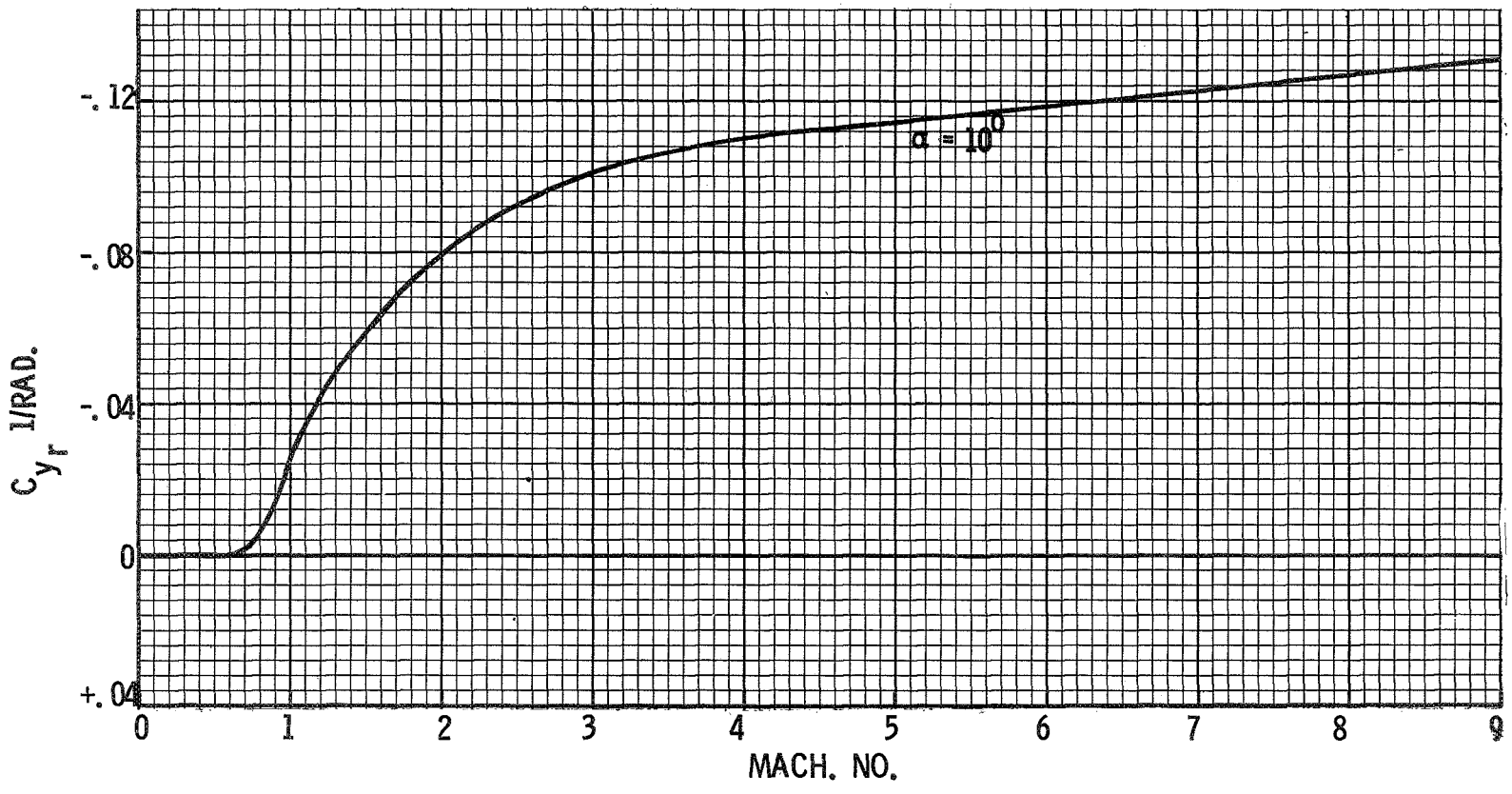


Figure B-22. Lockheed booster side force due to yaw rate.

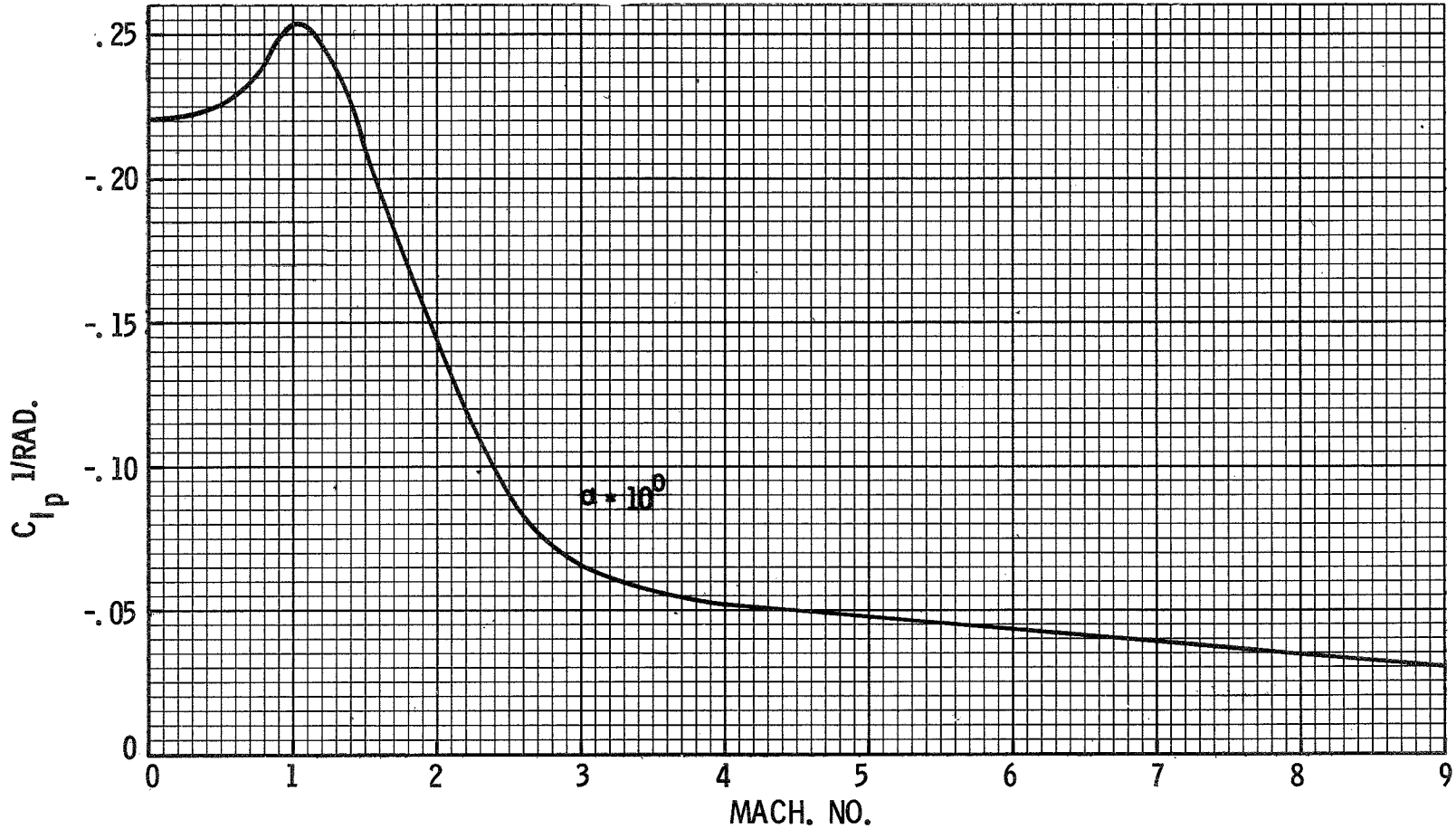


Figure B-23. Lockheed booster damping in roll.

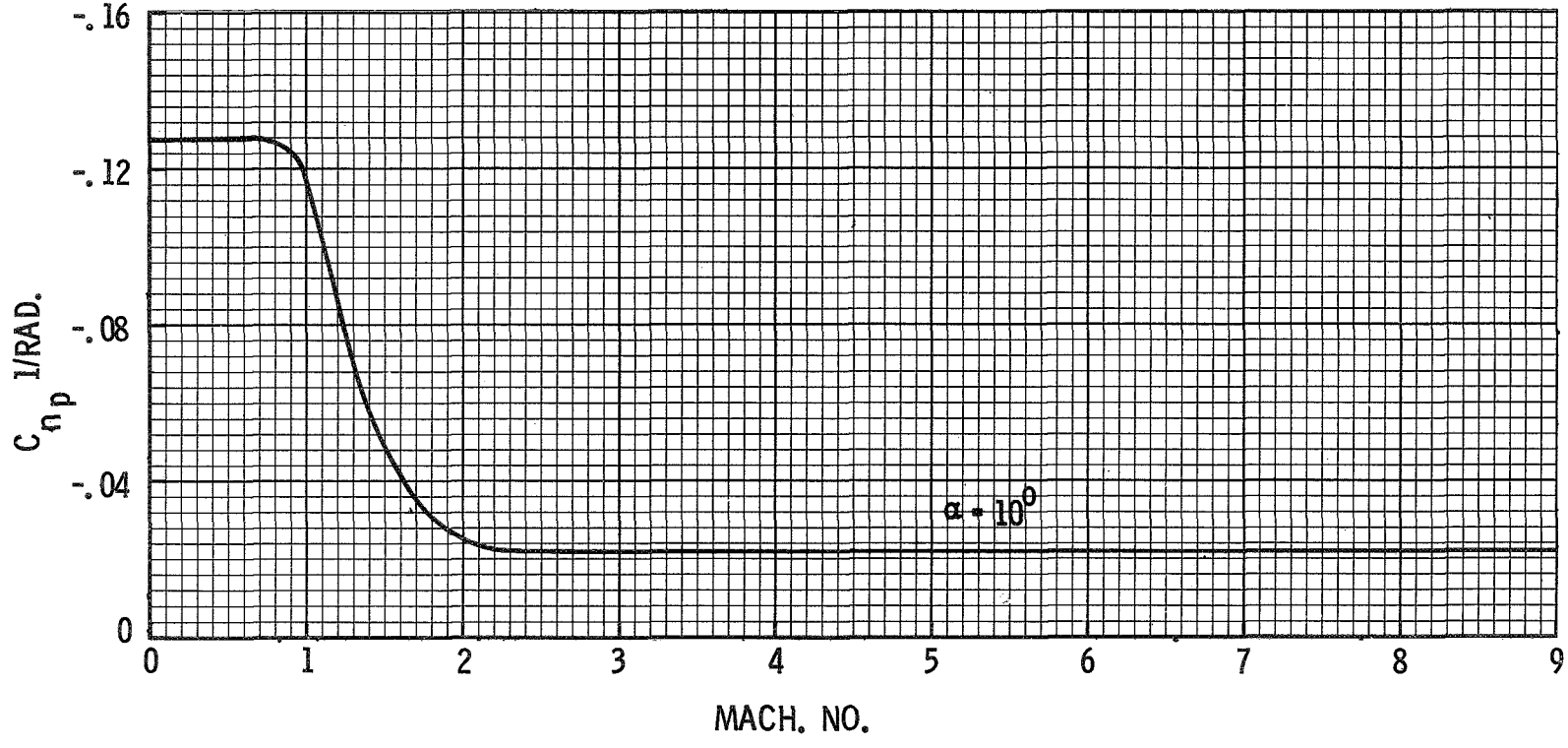


Figure B-24. Lockheed booster yawing moment due to roll rate.

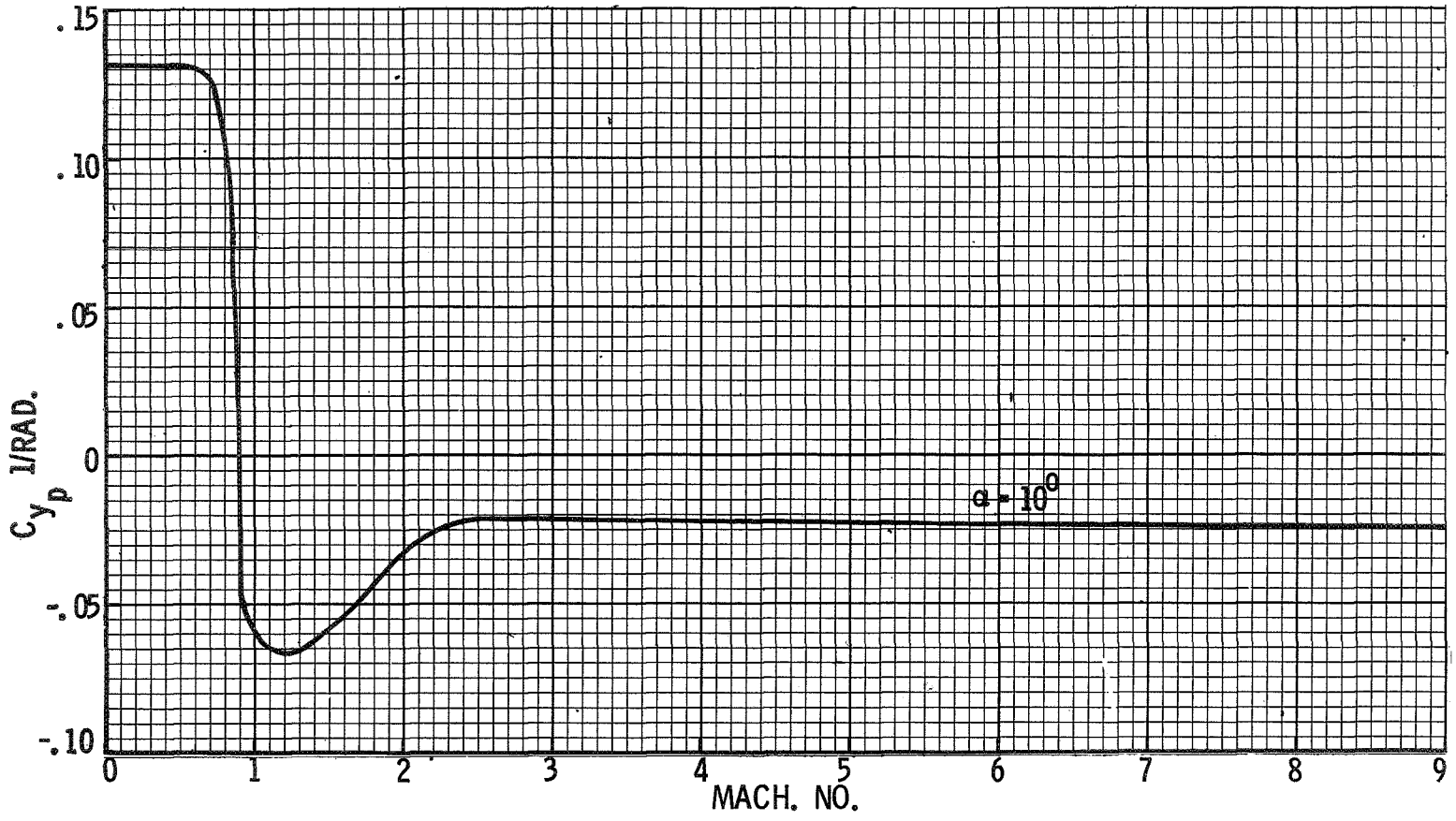


Figure B-25. Lockheed booster side force due to roll rate.

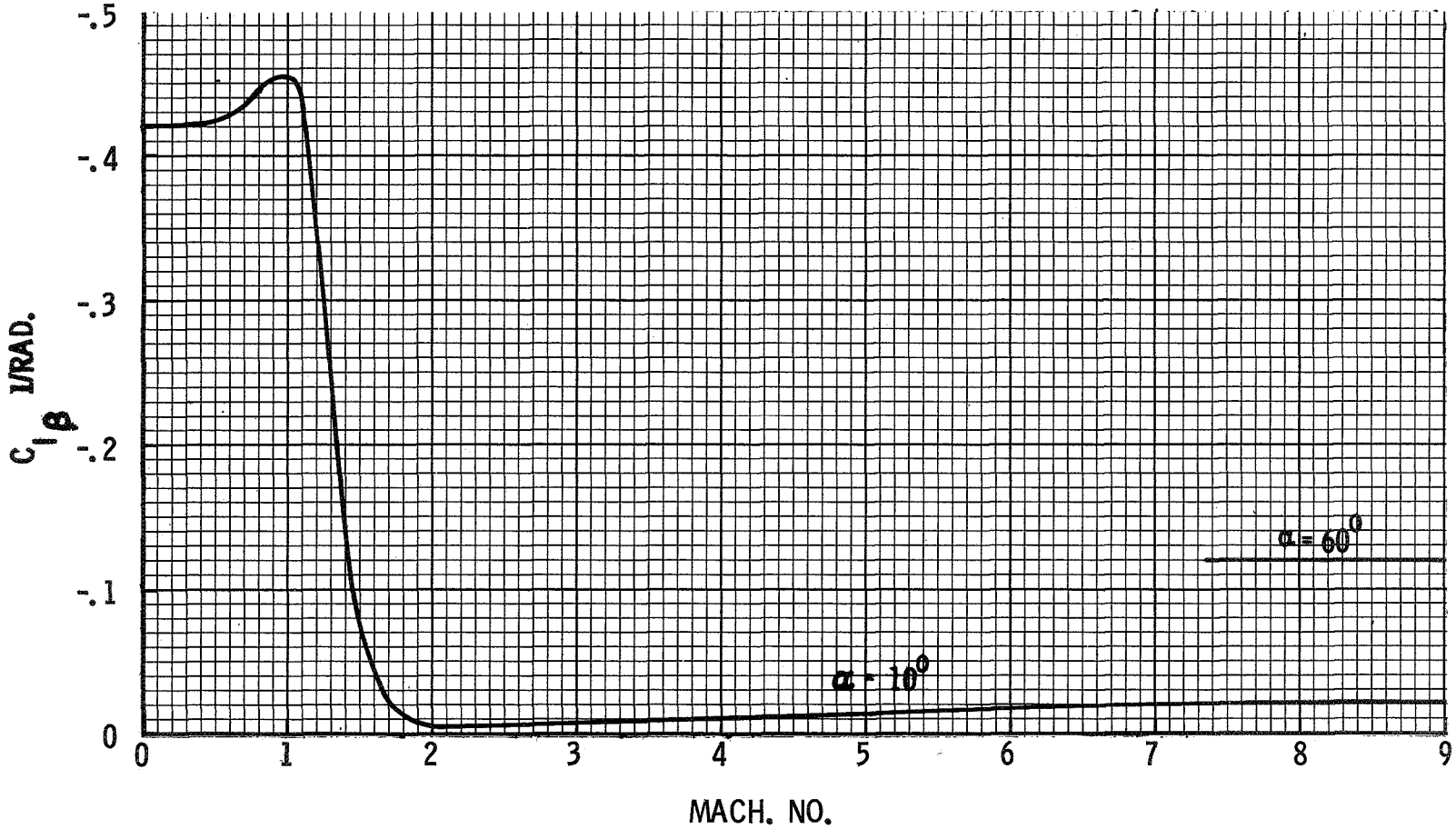


Figure B-26. Lockheed booster rolling moment due to sideslip.

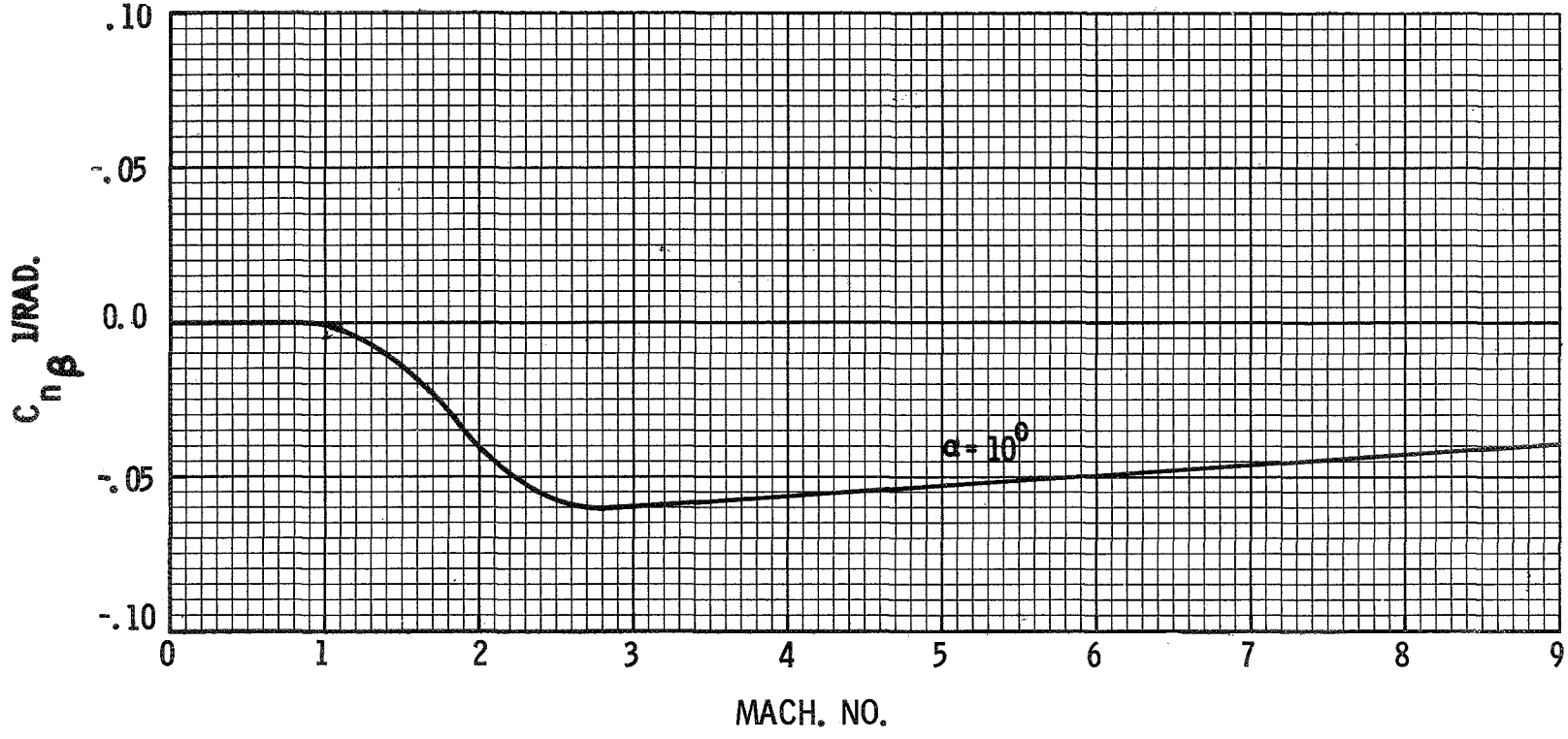


Figure B-27. Lockheed booster static directional stability.

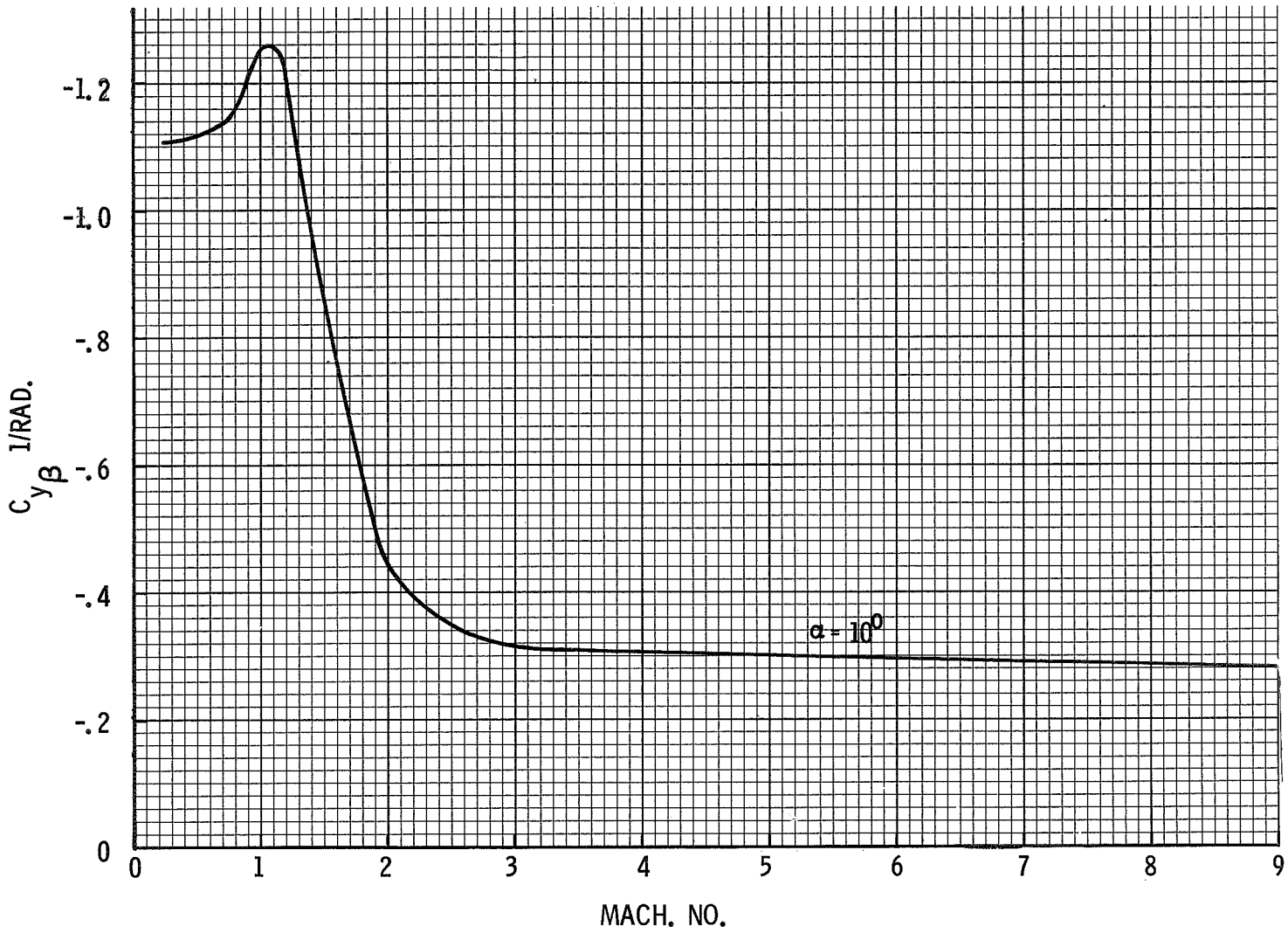


Figure B-28. Lockheed booster side force due to sideslip.

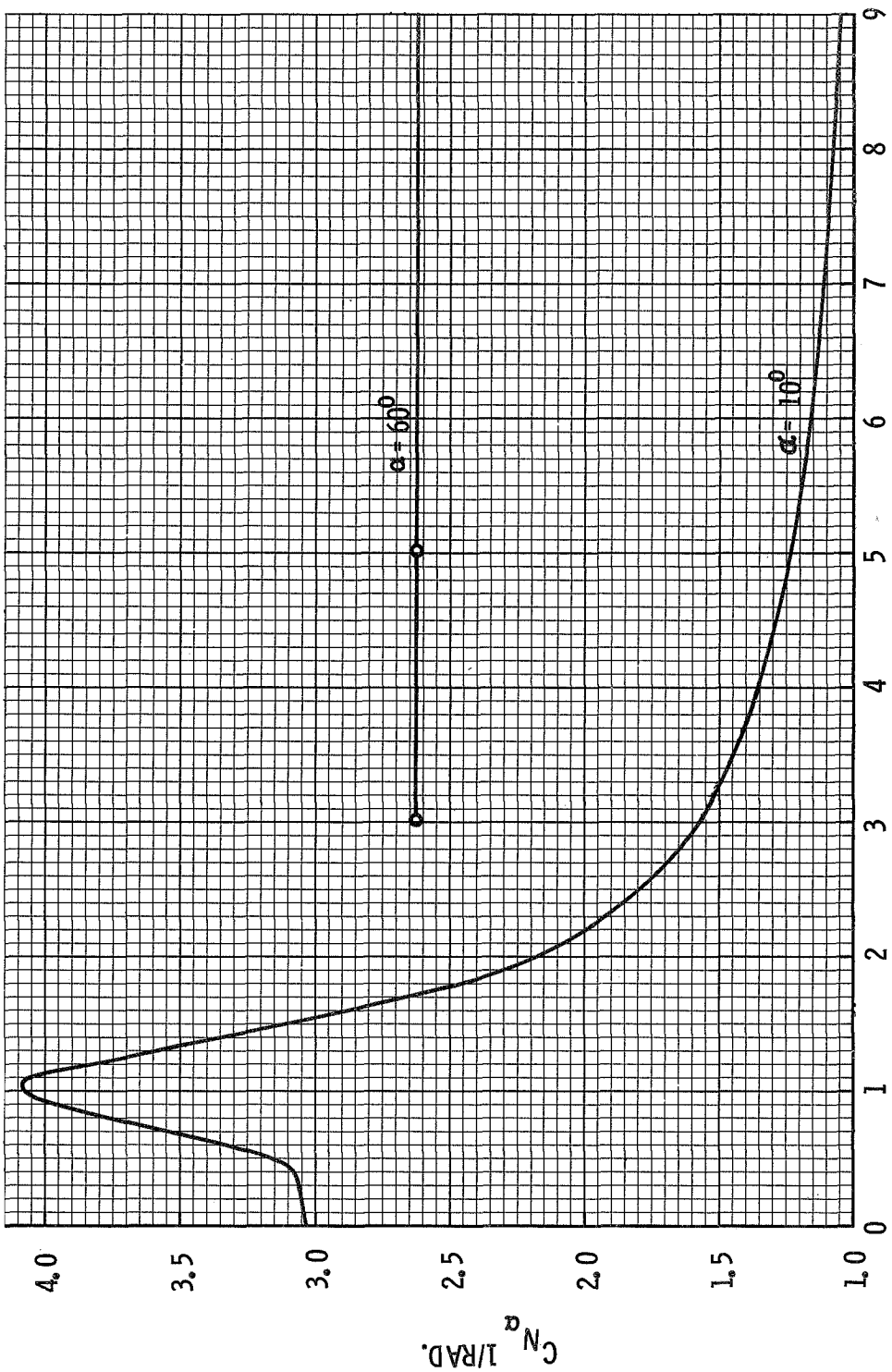


Figure B-29. Lockheed booster lift curve slope.

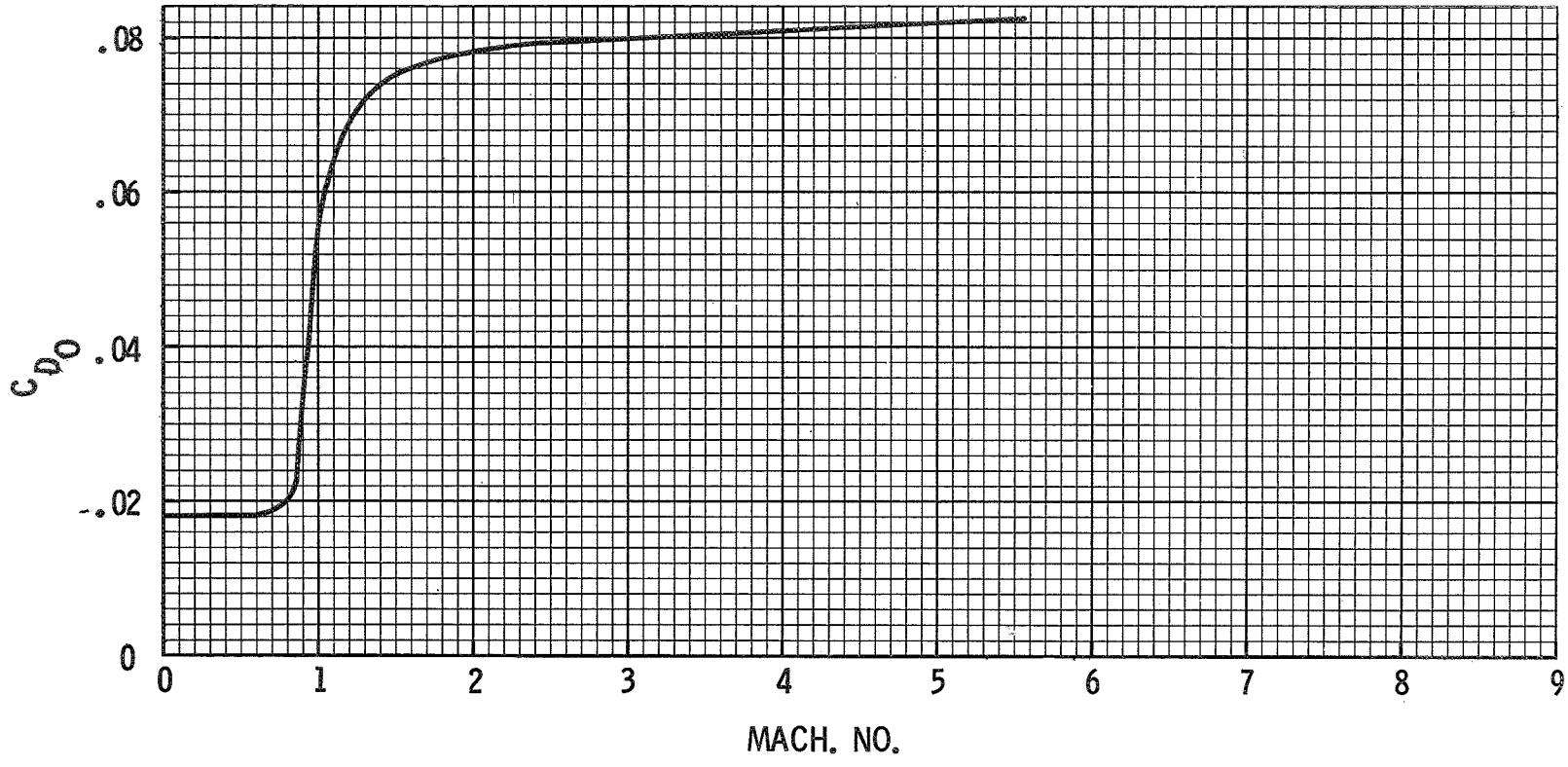


Figure B-30. Lockheed booster zero lift drag.

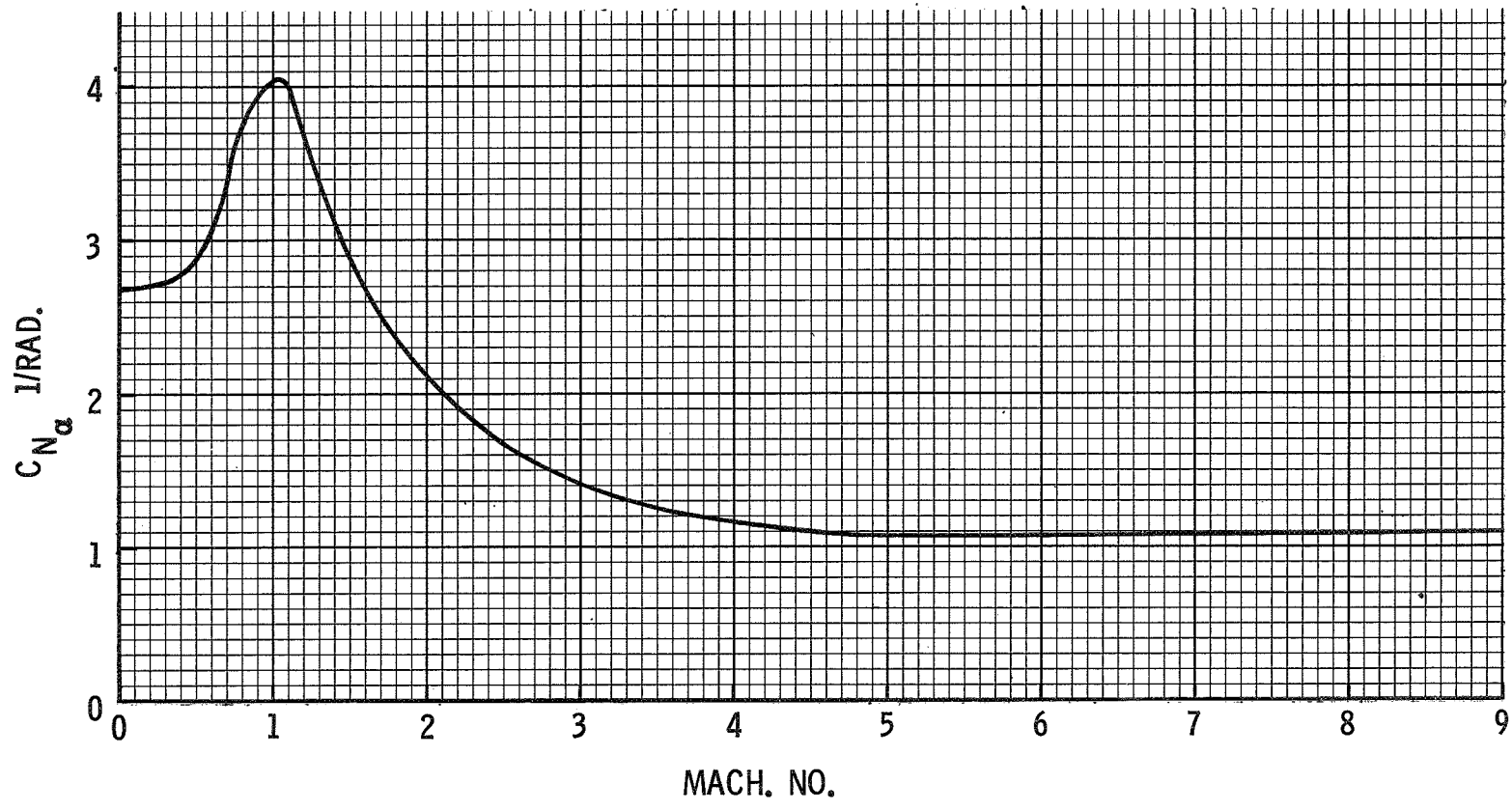


Figure B-31. Lockheed launch configuration lift curve slope.

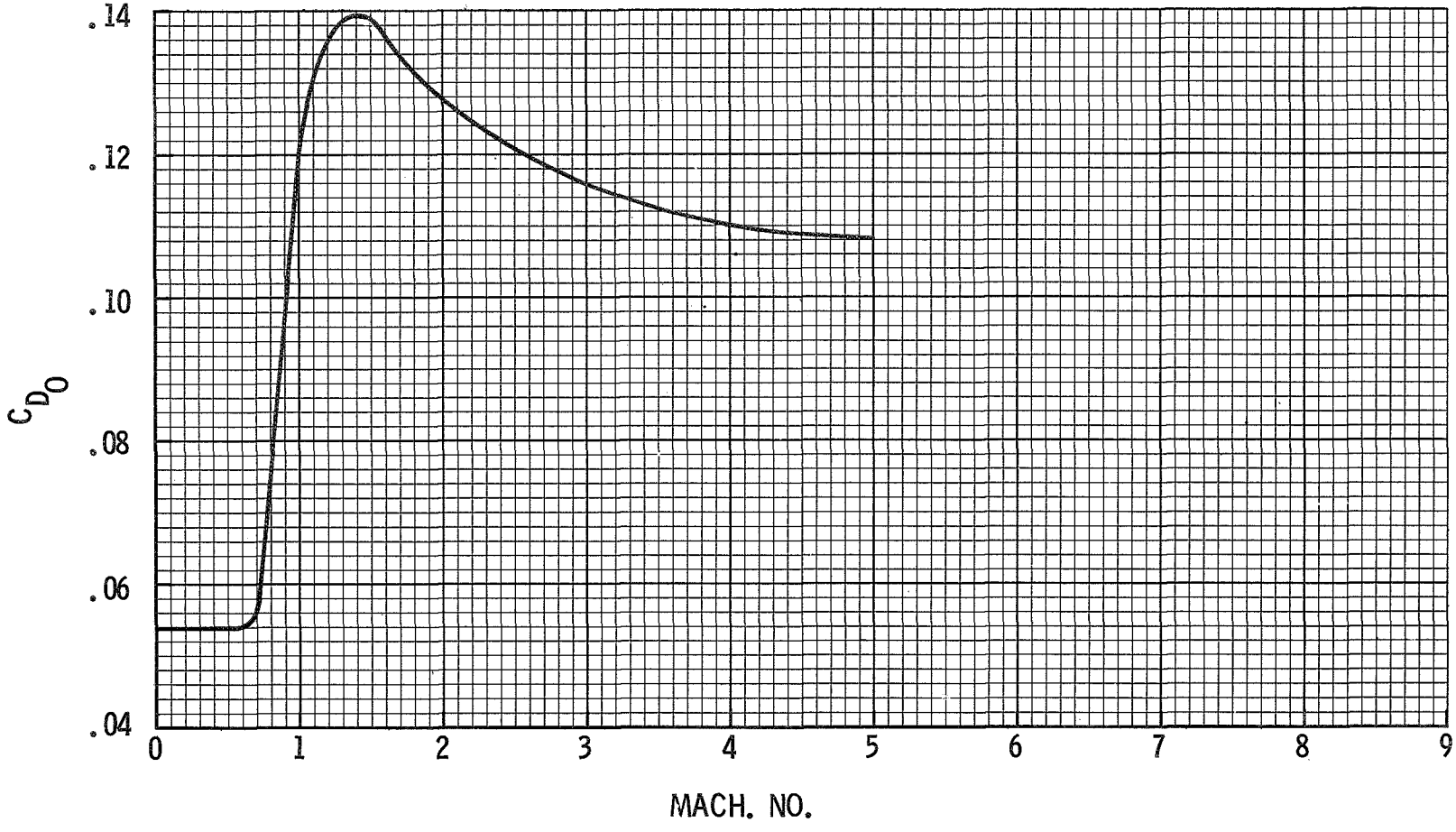


Figure B-32. Lockheed launch configuration zero lift drag.

APPENDIX C. DISPLAY TECHNOLOGY

Introduction

Display technology in recent years has begun to make extensive use of pictorial displays for information presentation. Pictorial displays can be defined as a technique whereby the relationship between many variables can be presented simultaneously in a visual form on a common frame of reference. Typical devices being used by the aircraft industry at this time are vertical situation displays (VSD), horizontal situation displays (HSD), heads-up-displays (HUD), electronic attitude director indicator (EADI), and multi-function displays (MFD). Some of the techniques being utilized to implement these devices are discussed in the following paragraphs.

Heads Up Display (HUD)

HUD combines the techniques employed in airborne gunsights with an electronic display to satisfy the compelling urge of an operator to view his environment during periods when his instruments must be monitored. Through techniques of this type a windshield display can be provided that allows a simultaneous view of the avionics displayed and the external environment unhampered by adverse weather conditions. A functional block diagram of a HUD is shown in Figure C-1.

Three methods of implementing the HUD are shown in Figures C-2, C-3, and C-4. They consist of reflective and refractive CRT systems and an electro-mechanical system. A brief comparison of the techniques is made in Table C-1.

Manufacturers of the different devices are as follows:

(1) CRT — Reflective Optics Device

General Electric

United Aircraft Corp., Norden Division

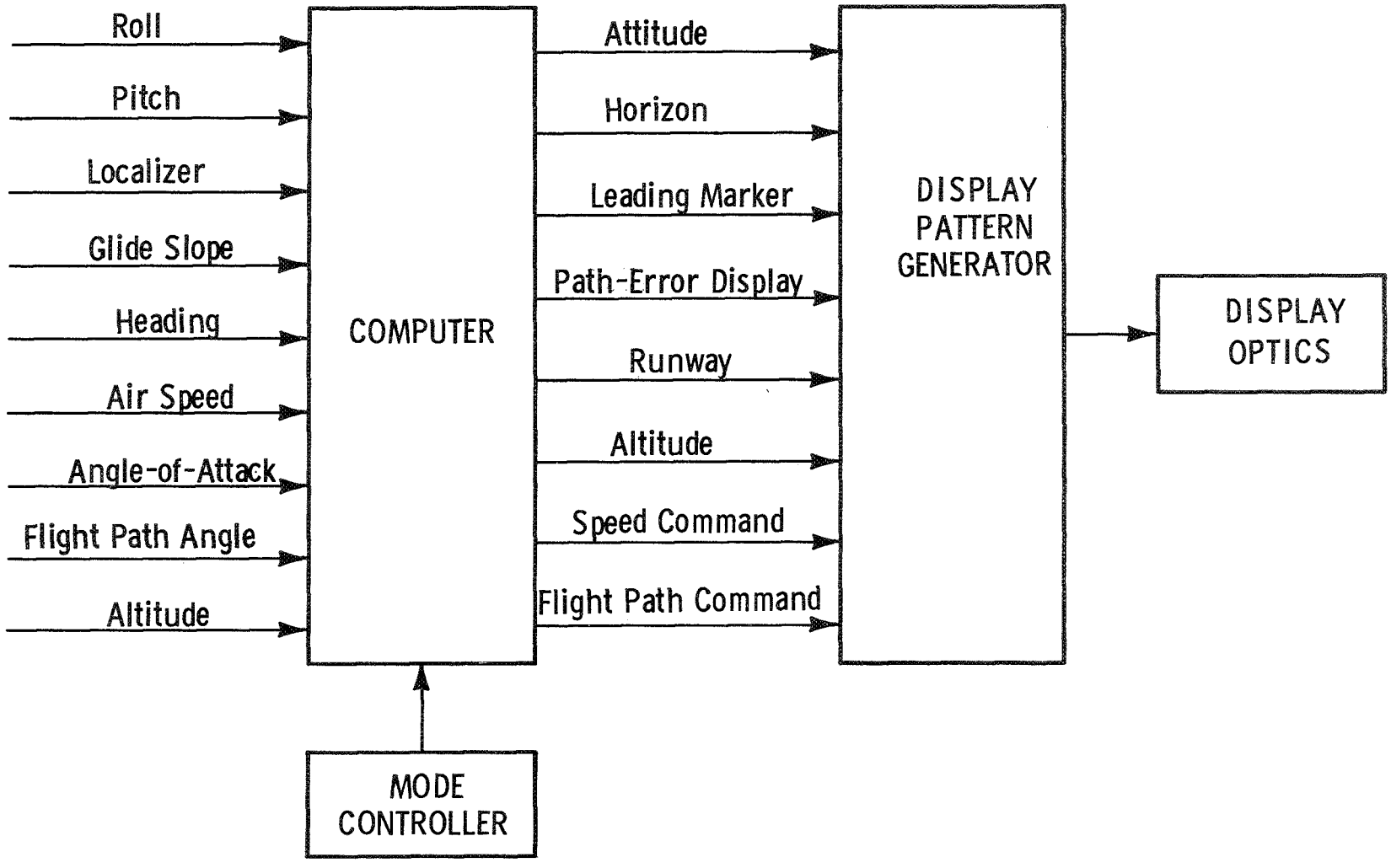


Figure C-1. HUD typical system block diagram.

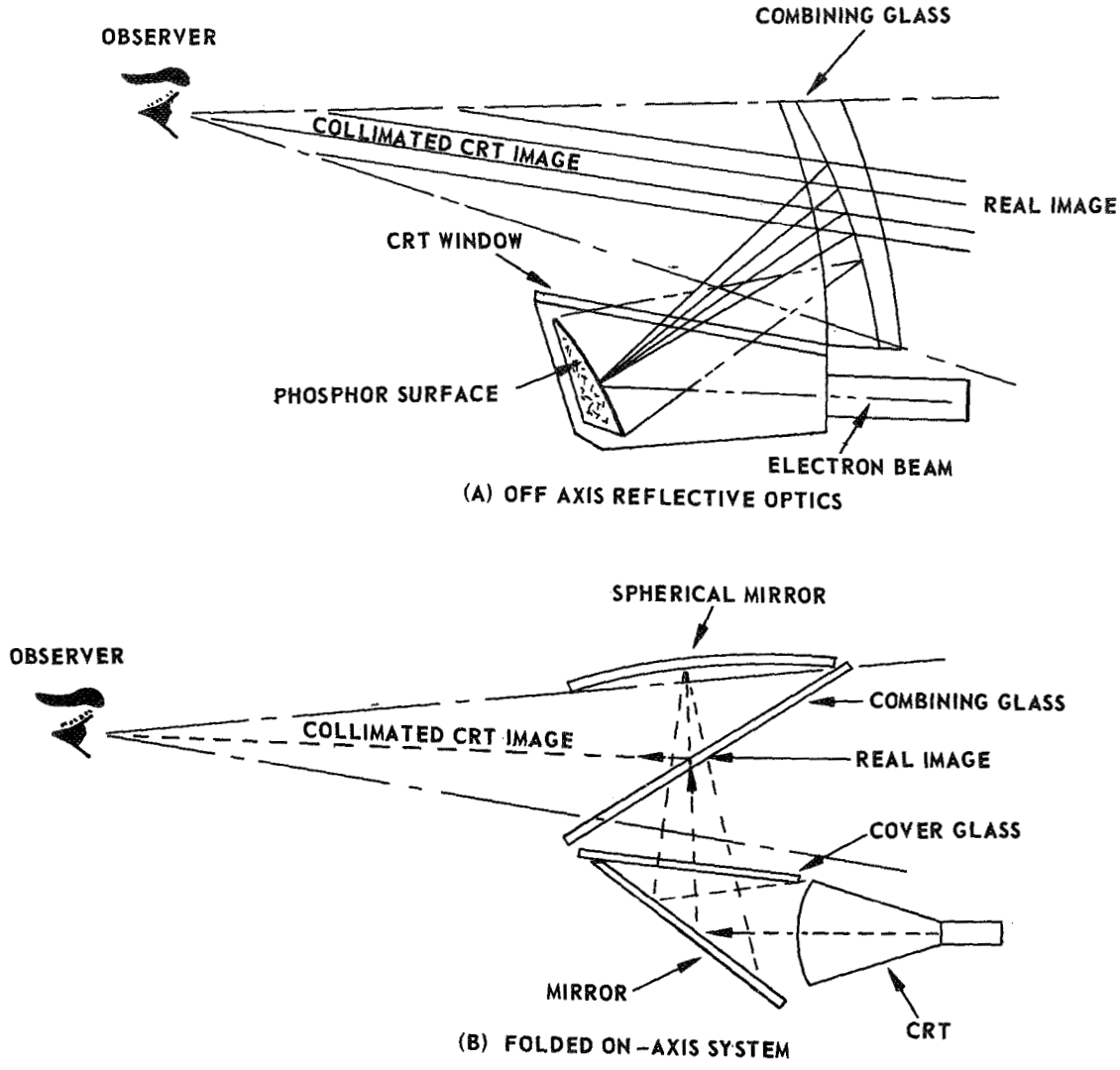


Figure C-2. CRT — Reflective optical systems.

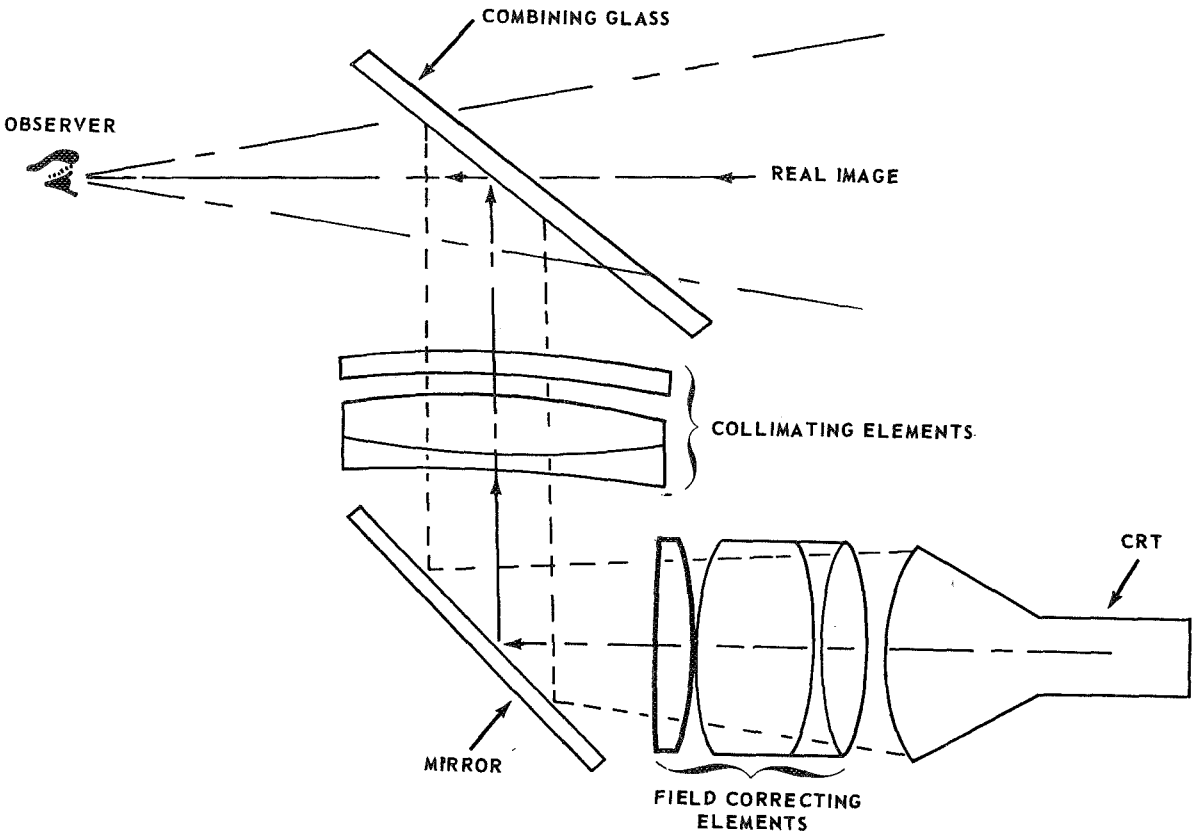


Figure C-3. CRT — refractive optics systems.

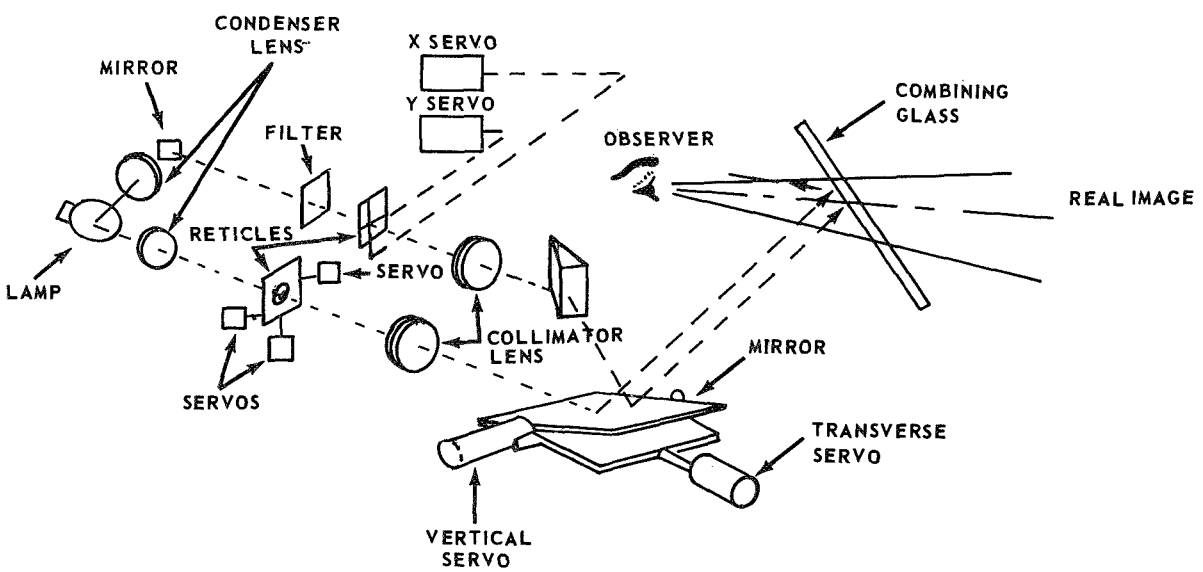


Figure C-4. Electromechanical system.

TABLE C-1. HUD IMPLEMENTATION COMPARISON

Type	Advantages	Disadvantages
CRT Reflective Optics	Instantaneous response	Large field of view (apparent infinite focus)
	Flexible symbology	
	Digital system (good data bus interface capabilities)	Viewer reflection (on-axis system)
		CRT EMI shielding (off-axis system)
CRT Refractive	Instantaneous response	Limited field-of-view (knothole effect)
	Flexible symbology	
	Experience (most widely used)	
	Digital system (good data bus interface capabilities)	
Electromechanical	Higher accuracy	Limited symbology
	Colored symbols	Dynamic response is limited
		Size and weight penalties
		Analog device (poor interface with digital data systems)

(2) CRT — Refractive Optics Devices

Conductron

Sperry Rand

Kaiser

Bendix

Others

(3) Electromechanical

Librascope (Linger — GPI).

The average parameters of commercial units presently available are:

(1) Weight — 42 pounds

(2) Power — 175 watts

(3) Volume — approximately 2000 cubic inches.

If incorporated into the shuttle display system, the HUD electronics would be integrated into the shuttle display electronics, thereby reducing these figures to values only slightly higher than those of the display itself. A typical electronics block diagram of a HUD is shown in Figure C-5.

Regardless of implementation techniques, the optical system is an essential element of the display. The display data (symbols, etc.), formed by a reticle and/or CRT, must be combined and projected in a manner such that the images appear as they would at infinity; this is accomplished by the optical collimation system. The light rays emanating from each source are reflected from a combining mirror located in the window area. As the observer looks through the combining glass at the external environment, the images appear ahead of the observer superimposed upon his view.

Size of display is dictated by human factors and vehicle characteristics. An experimental all-weather aircraft nominally requires a 25-degree angular field of view. This is equivalent to looking through a 12.5-inch porthole from a distance of 30 inches.

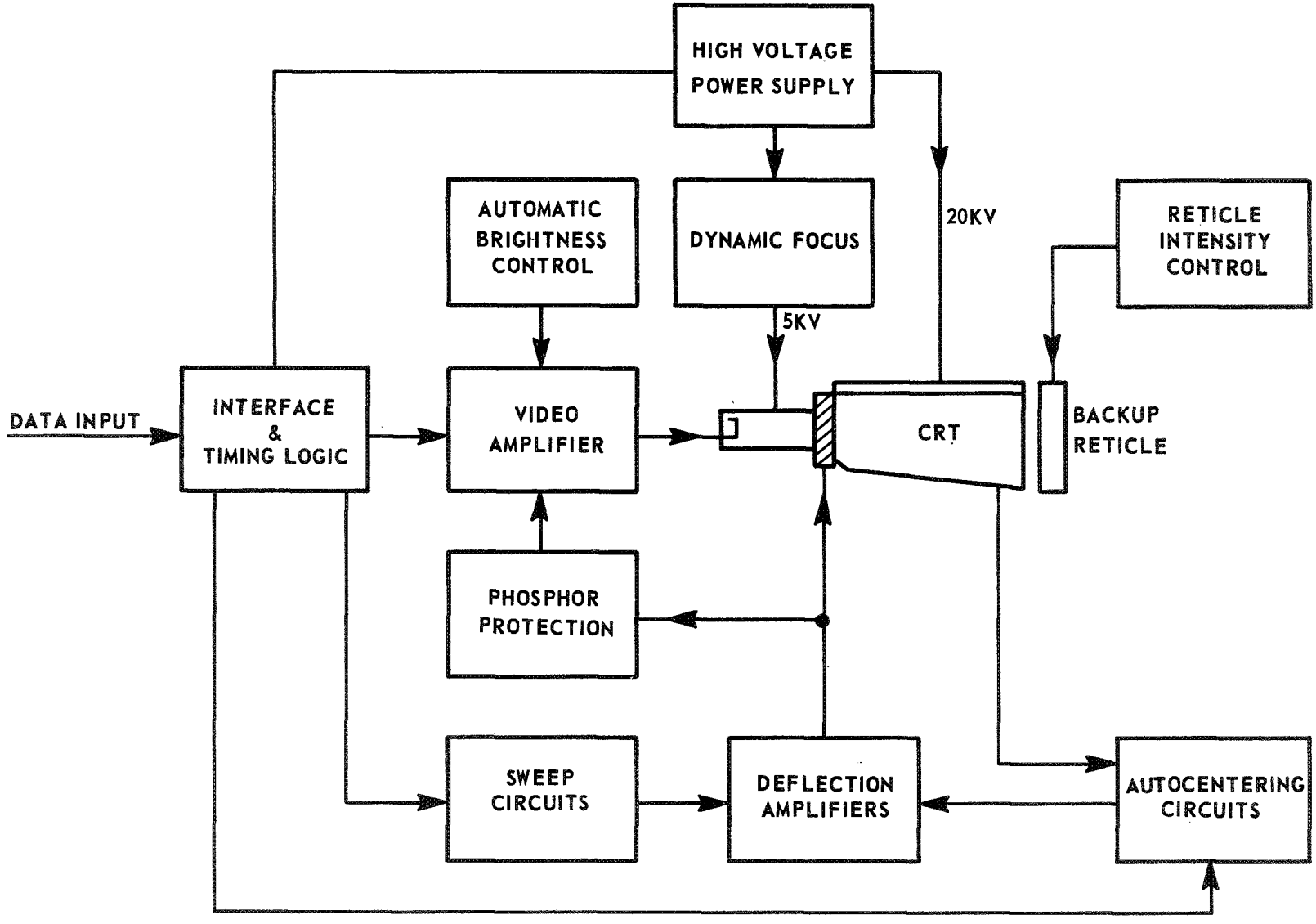


Figure C-5. Typical HUD electronics block diagram.

There are two basic types of optical systems utilized for the HUD: extended pupil and gunsight. The first has the advantage that the observer notices no material change in the displayed data as long as his head motion is limited. Movement beyond this limit results in disappearance of the display data. The second method exhibits a knothole effect. Limited head motion results in disappearance of varying portions of the data display.

The distinguishing feature of the extended pupil system is the curved combiner. The ideal combiner would be a sphere with the observer at its center. As the optics vary from this ideal, distortion occurs. Also, off-axis projection is required, resulting in additional distortion errors. These errors can be overcome, but they require additional electronic and/or optical correction features.

Display symbology and amount of data displayed varies from system to system. Figure C-6 illustrates typical display formats for a HUD in four modes of operation. These examples are for aircraft modes of operation; however, similar formats suitable for spacecraft operation can be provided by proper changes in symbol generation and formatter design.

Horizontal Situation Display

The historic use of the HSD has been for plan position indication which is useful for aircraft navigation or geographic orientation. In addition, the display serves as a display for radar and TV data, alphanumeric instructions, and a variety of symbols depending on the particular mission phase. The typical HSD combines on a single screen a moving pictorial map, radio navigation data, electronically generated symbology overlays, and radar and TV information. The need for the usual radio deviation, bearing, distance, heading, and course indicators is eliminated by combination of the data on the screen of this single instrument.

Figures C-7 through C-11 depict four methods of implementing the HSD. A comparison of these methods is made in Table C-2. Of these, the optronic display has the most desirable features. Figure C-12 shows the optical system for a typical optronic display HSD. One added feature of this system, not shown here, is the capability to use one of the rear ports to photograph the display for data retention and post flight evaluation.

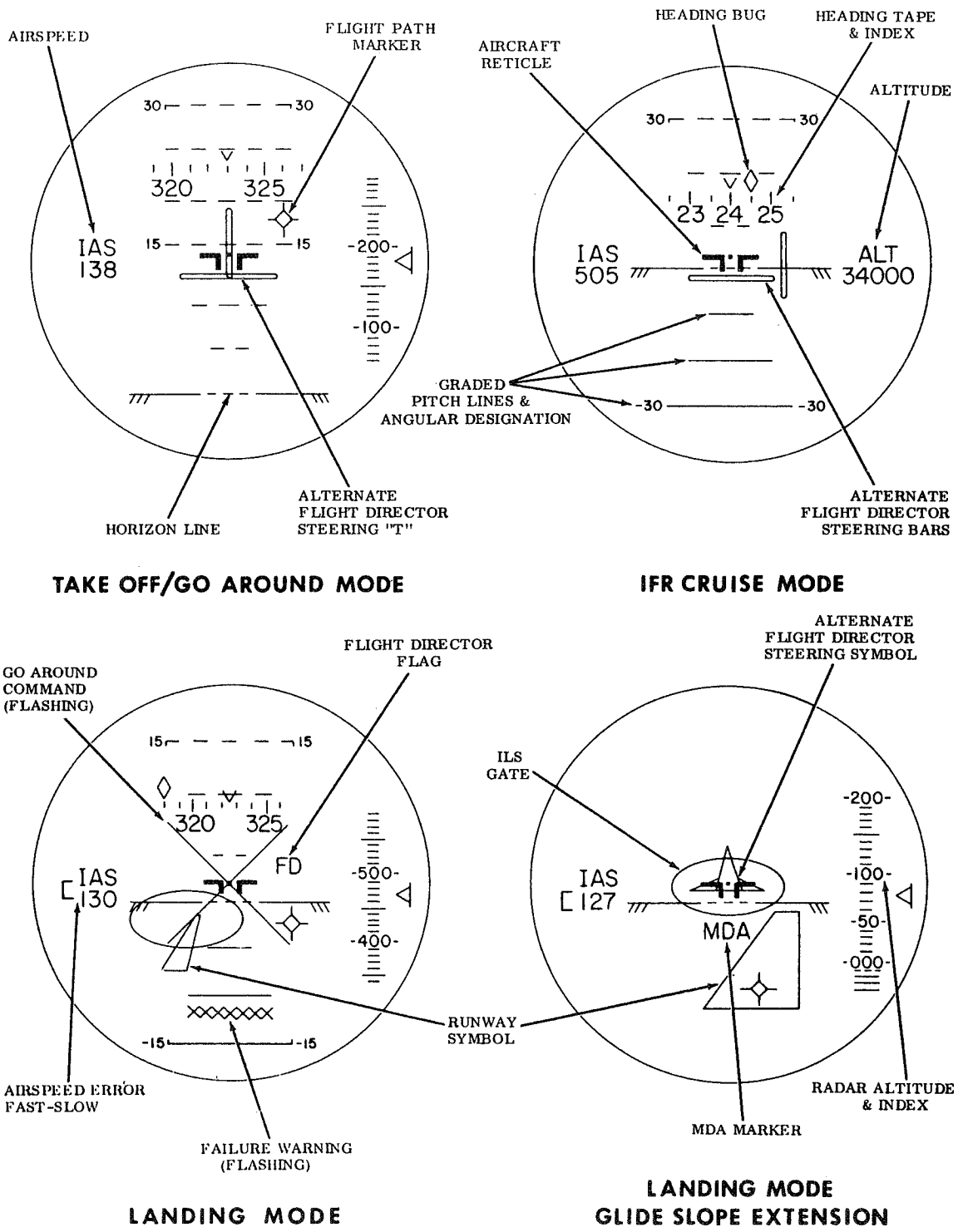


Figure C-6. Typical aircraft HUD formats.

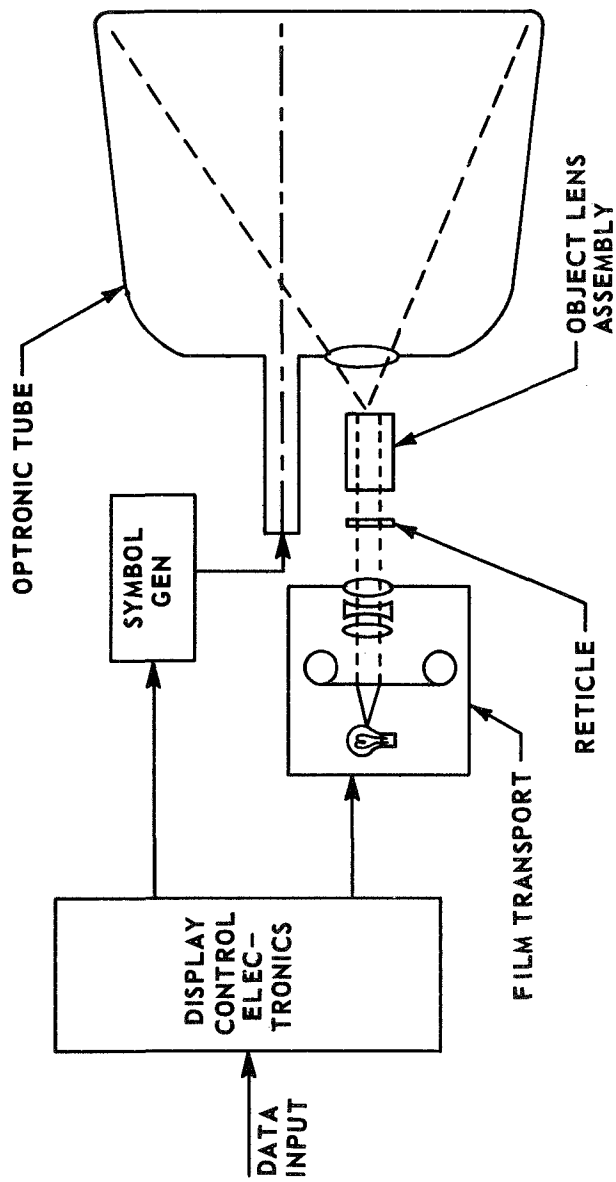


Figure C-7. Optronic (rear projection CRT) display.

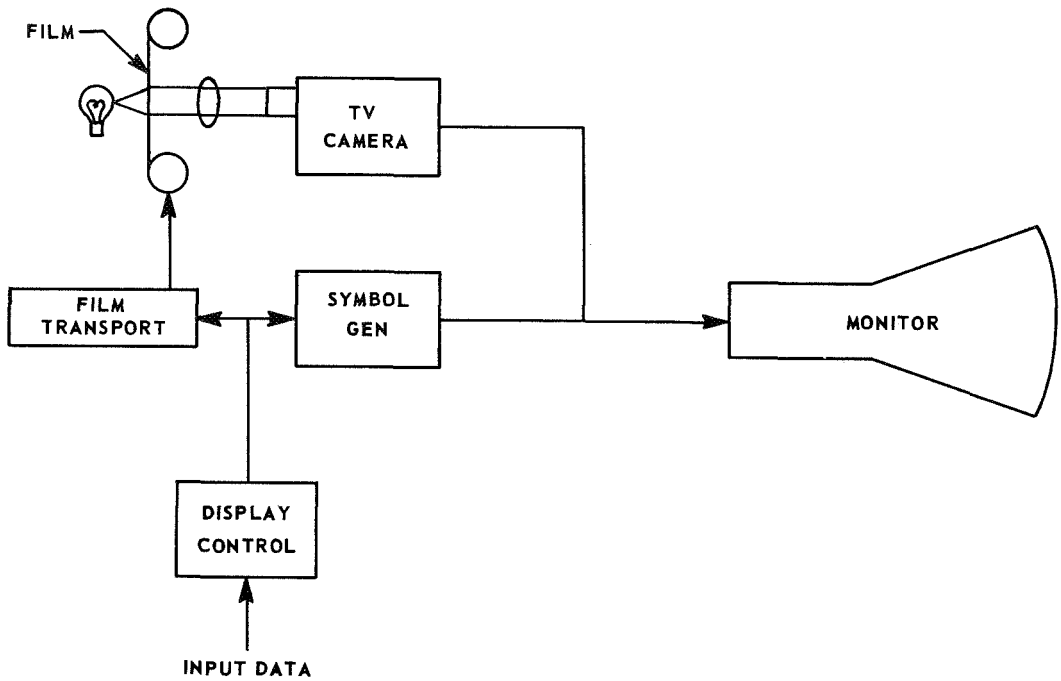


Figure C-8. Remote TV system.

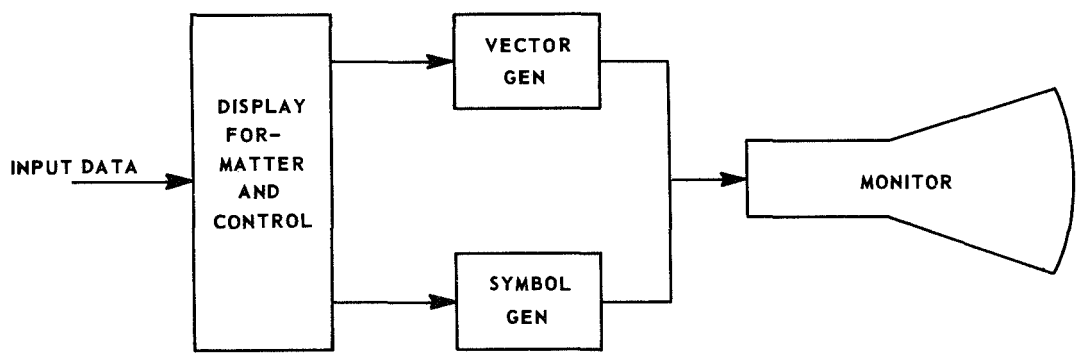


Figure C-9. Electronic display.

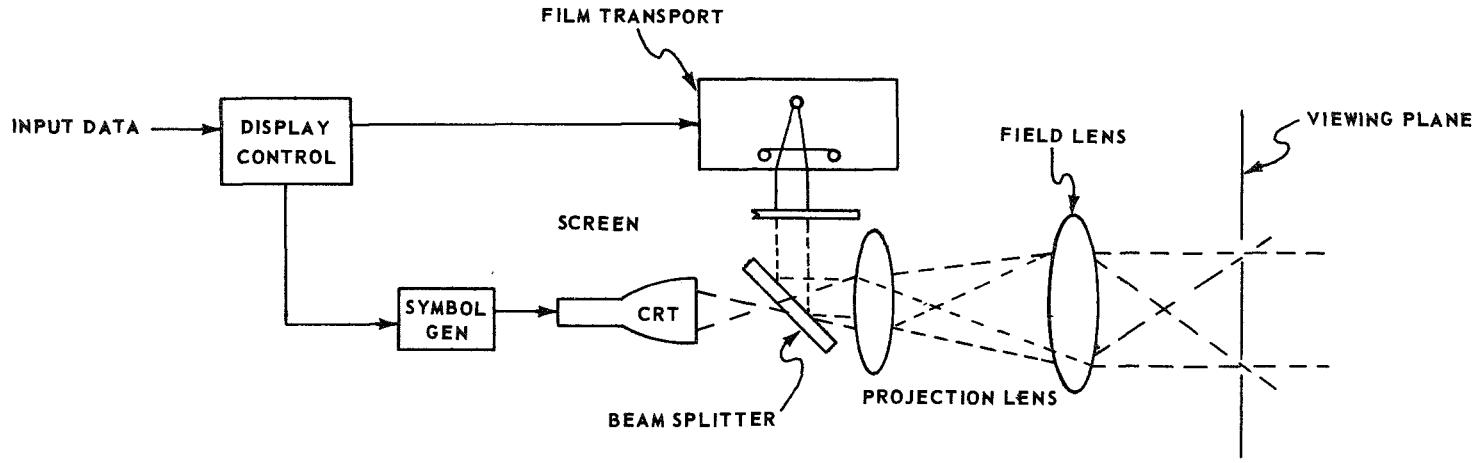


Figure 10. Optical — CRT display.

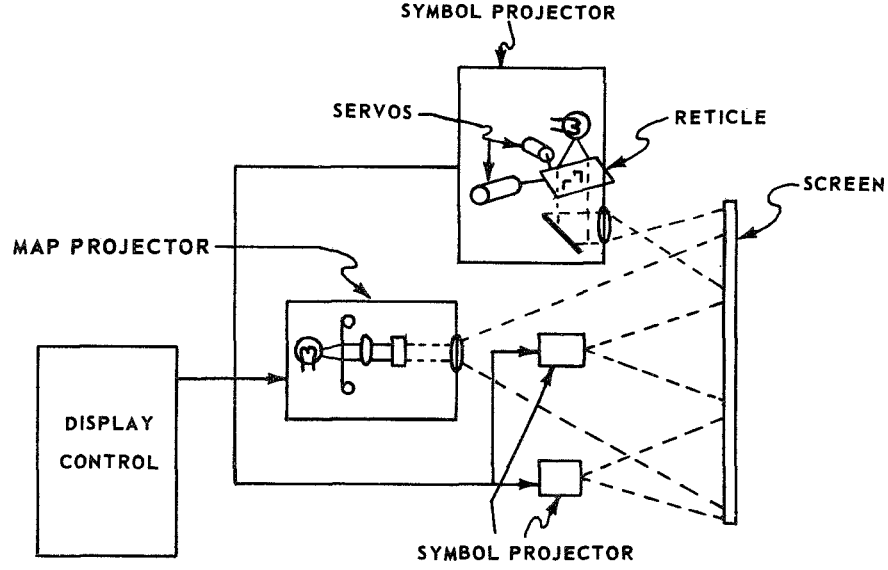


Figure C-11. Optical display.

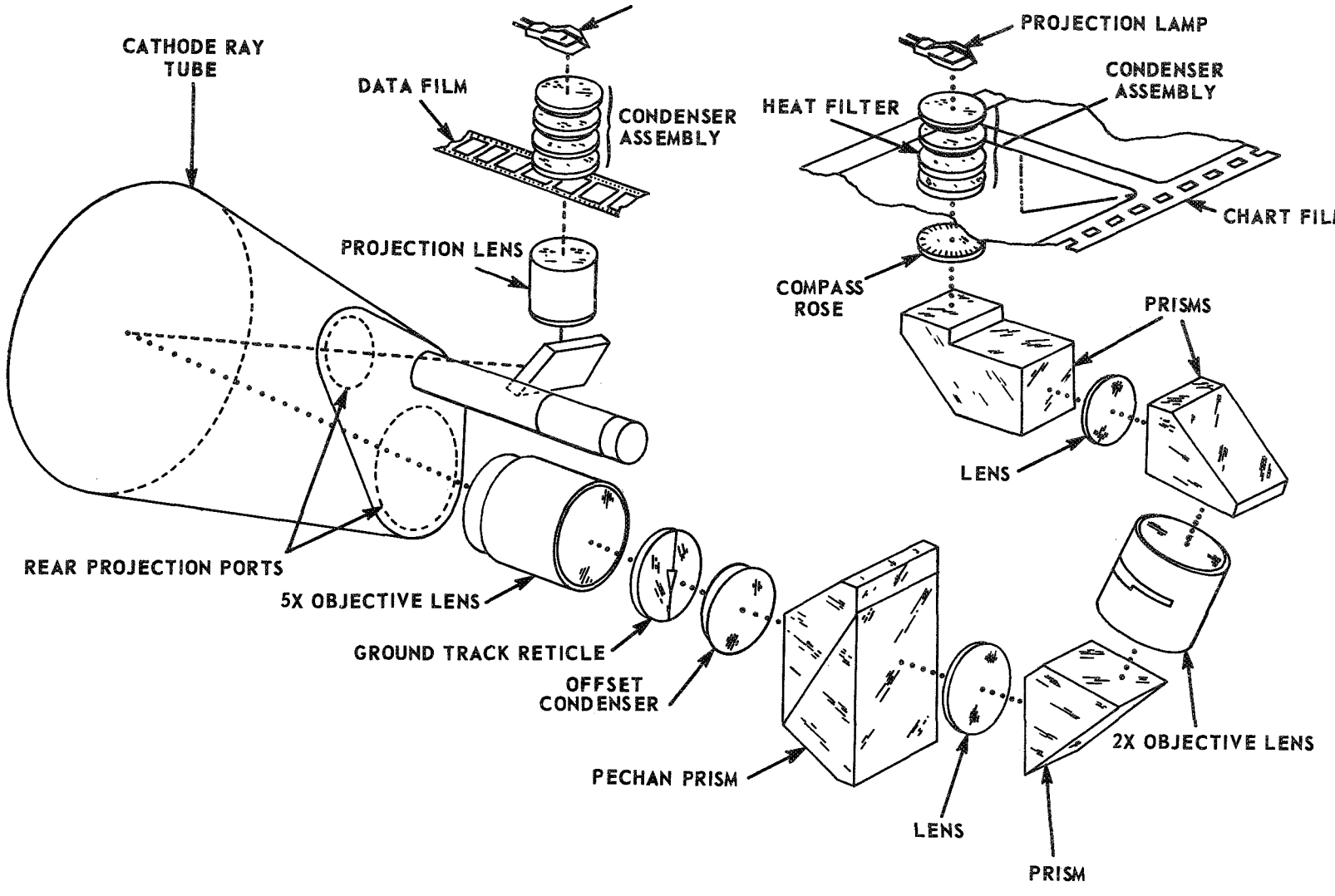


Figure C-12. Typical HSD optics for rear projection CRT's.

TABLE C-2. HSD IMPLEMENTATION COMPARISON

Type	Advantages	Disadvantages
Optronic	Symbol generation	
	Map and data generation	
	Flexibility	
Remote TV	Accuracy	Cost
	Small size	Poor map resolution
	Flexibility	
Electronic	Symbol generation	Poor map resolution
	Brightness	Severe computer requirement for map generation
	Size	
	Weight	
	Maintenance	
	System flexibility	
Combined Optical CRT	Symbol generation	Poor symbol resolution
	Map generation	
	Map resolution	
	Accuracy	
Optical	Resolution	Limited symbol generation
	Brightness	
	Accuracy	
	Cost	

Multimode Displays

In addition to the HSD, other referenced multimode displays are the MFD and EADI. Functional block diagrams of the two displays are shown in Figures 7-5 and C-13, respectively. The two most desirable techniques for implementing these displays are shown in Figures C-7 and C-9. Although neither of the systems requires rear projection capability, the optronic display has some advantages. In the case of the MFD, graphs and other predetermined information can be stored on film, thus reducing computer memory and display generation requirements. To achieve display commonality and complete redundancy of displays, the rear projection tube is required for all CRT displays.

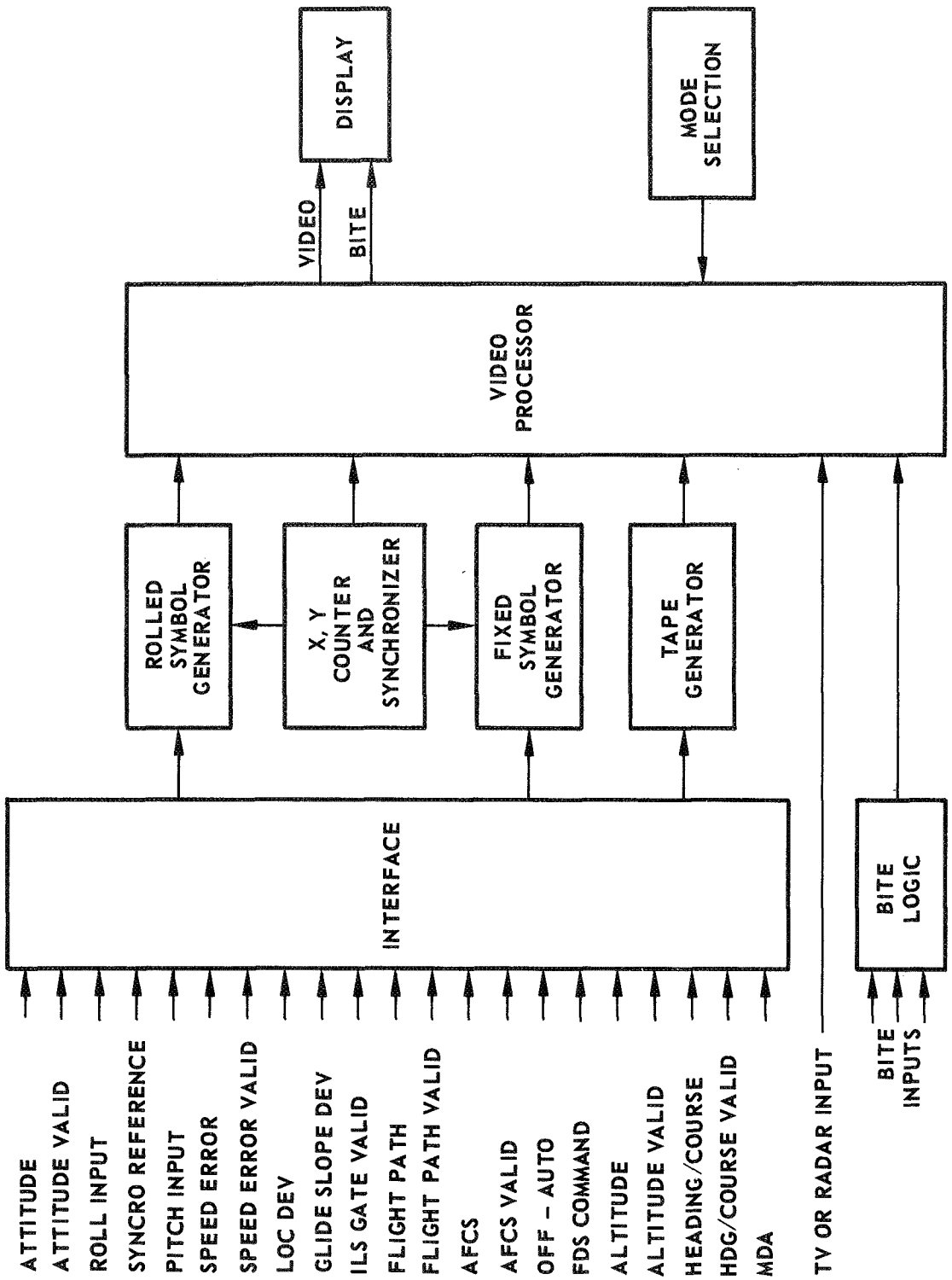


Figure C-13. EADI functional block diagram.

APPENDIX D. STRUCTURAL/MECHANICAL, PROPULSION, AND ENVIRONMENTAL CONTROL SUBSYSTEM BLOCK DIAGRAMS

Appendix D contains information on the nonavionics systems of the shuttle vehicle. Considerable information on these subsystems was generated by various contractor studies as well as by in-house NASA investigations. This reference material was used to determine the necessary background and interface information to permit the IAS design to be pursued. Basically, requirements and constraints imposed on the IAS by these various subsystems were identified and utilized where necessary in making the studies and investigations of the respective IAS subsystems.

An example of the type of information available is contained in Figures D-1.0 through D-4.5. These figures are first-level block diagrams of the structural/mechanical (Figs. D-1.0 through D-1.10), propulsion (Figs. D-2.0 through D-2.10), and environmental control (Figs. D-4.0 through D-4.5) subsystems of the shuttle vehicle. These basic diagrams were extracted from a study by Lockheed Missiles and Space Co., NASA Contract NAS9-9206. Modifications to the Lockheed drawings were made, where necessary, to remove implications of a specific vehicle configuration.

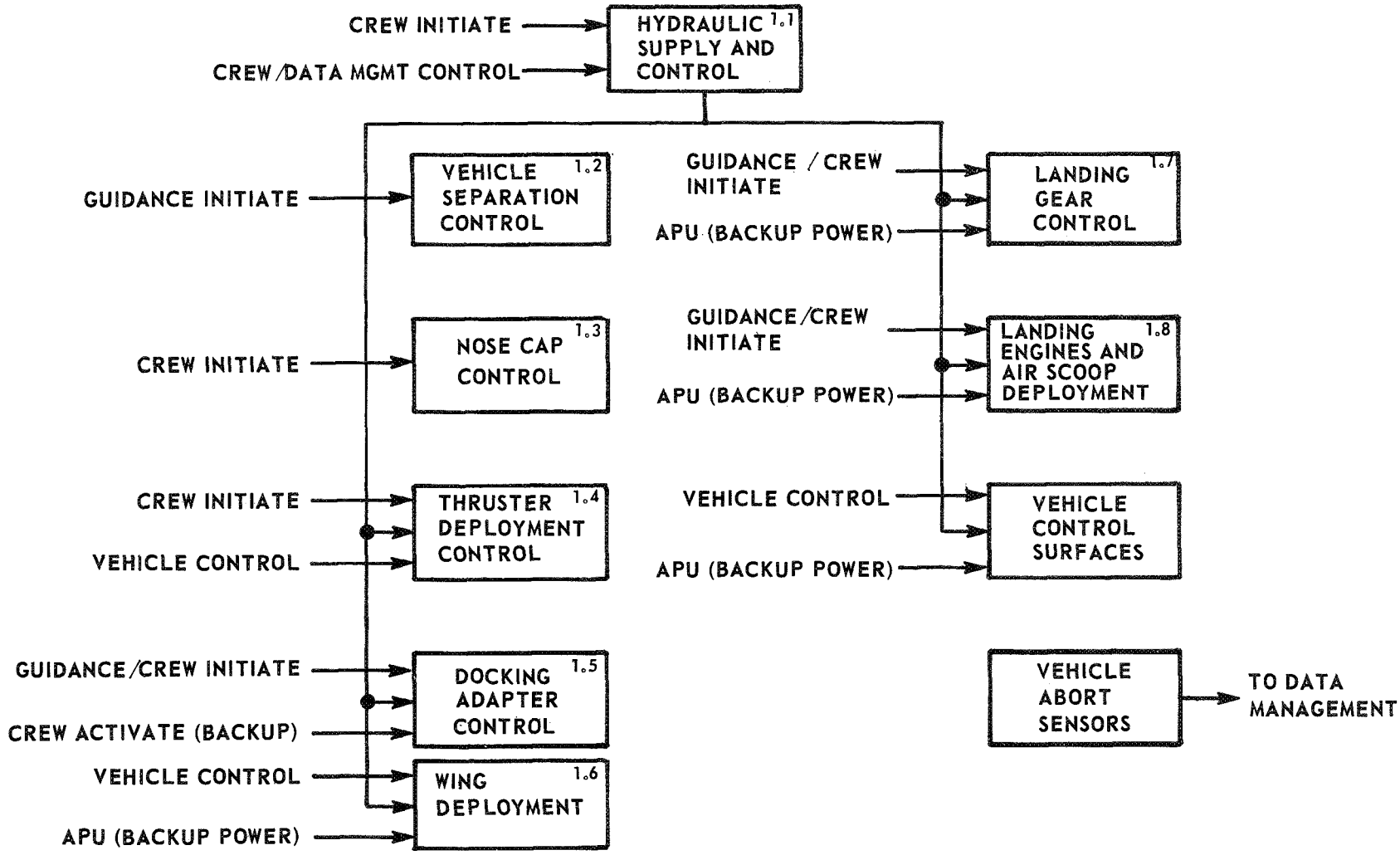
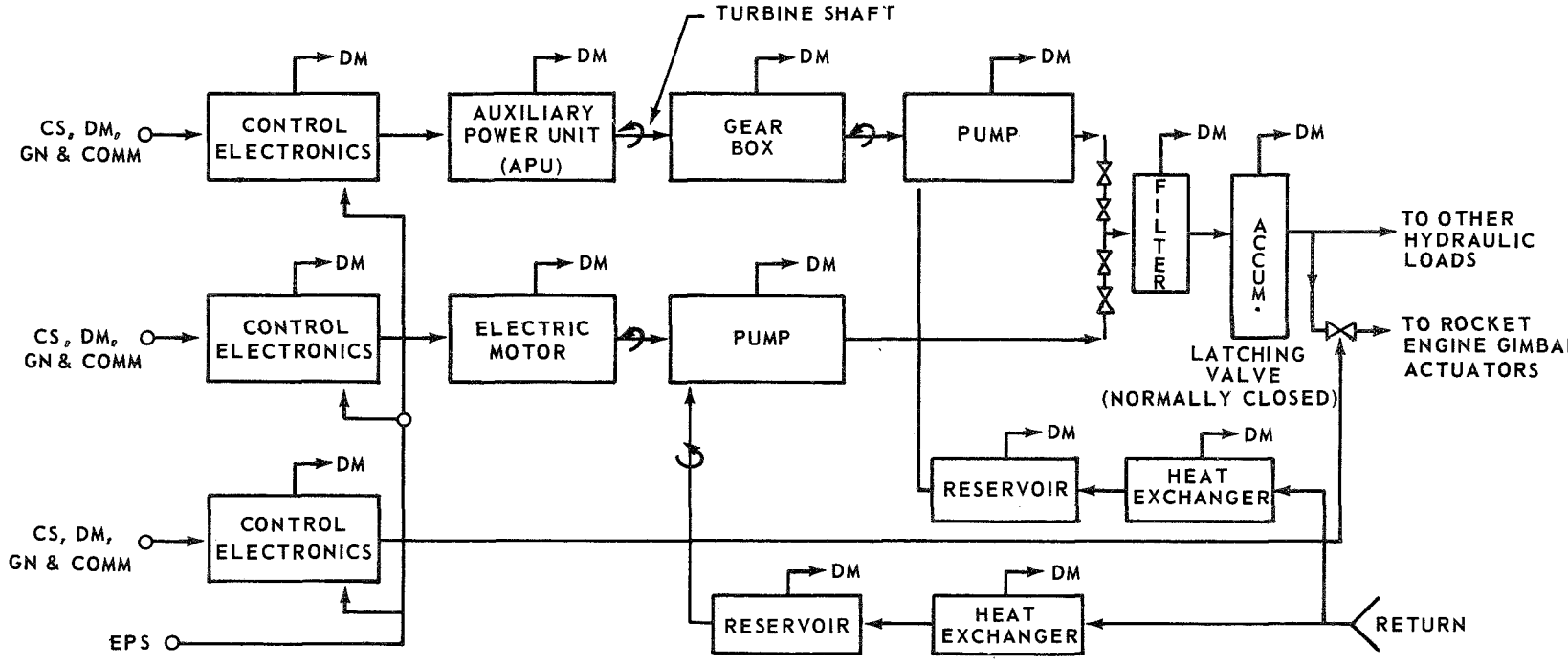


Figure D-1.0. Structural/mechanical.



NOTE:

1. EPS = ELECTRICAL POWER SYSTEM
2. DM = DATA MANAGEMENT
3. GN = GUIDANCE/NAVIGATION
4. COMM = COMMUNICATIONS
5. CS = CREW STATION

Figure D-1.1. Hydraulic supply and control.

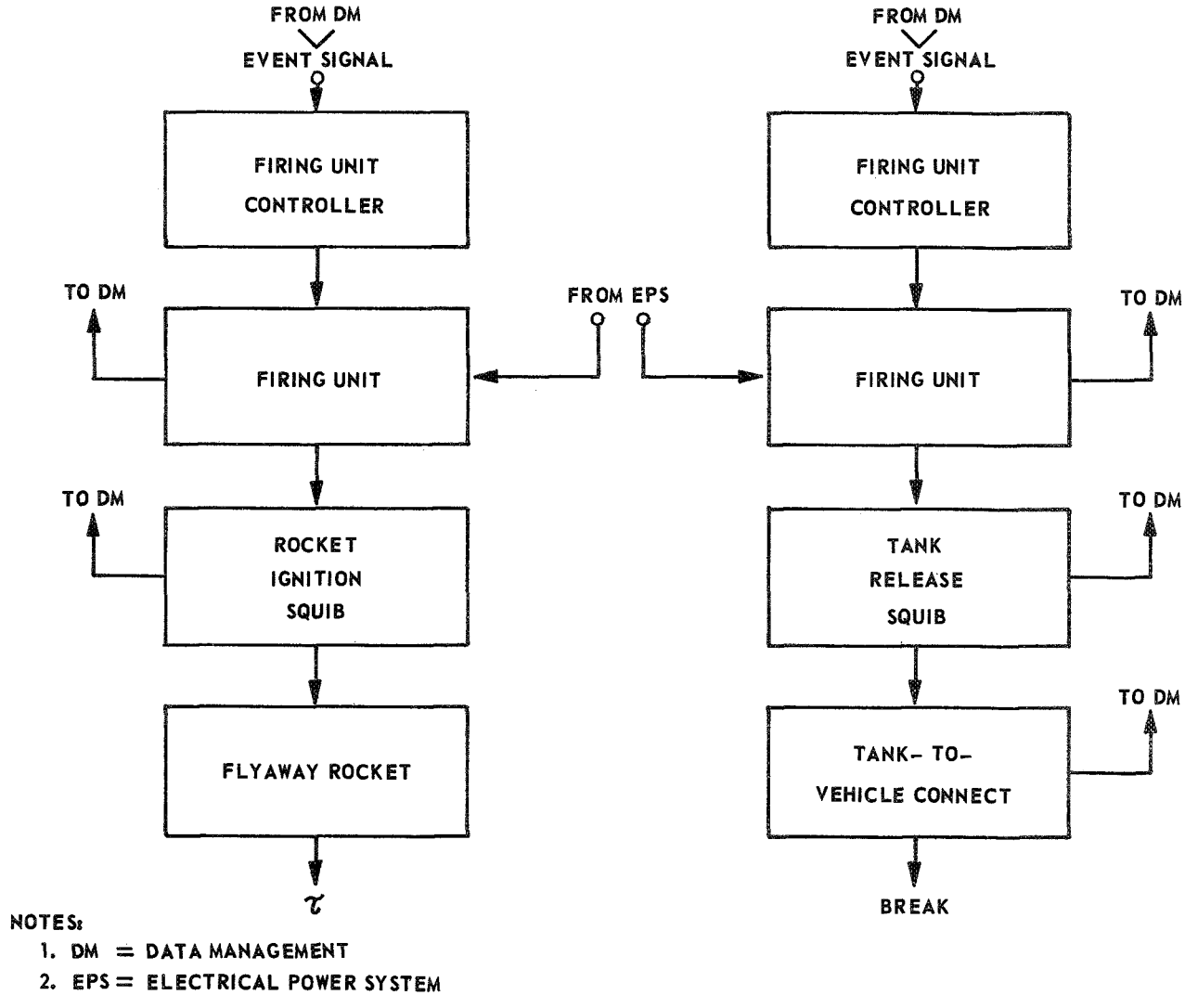
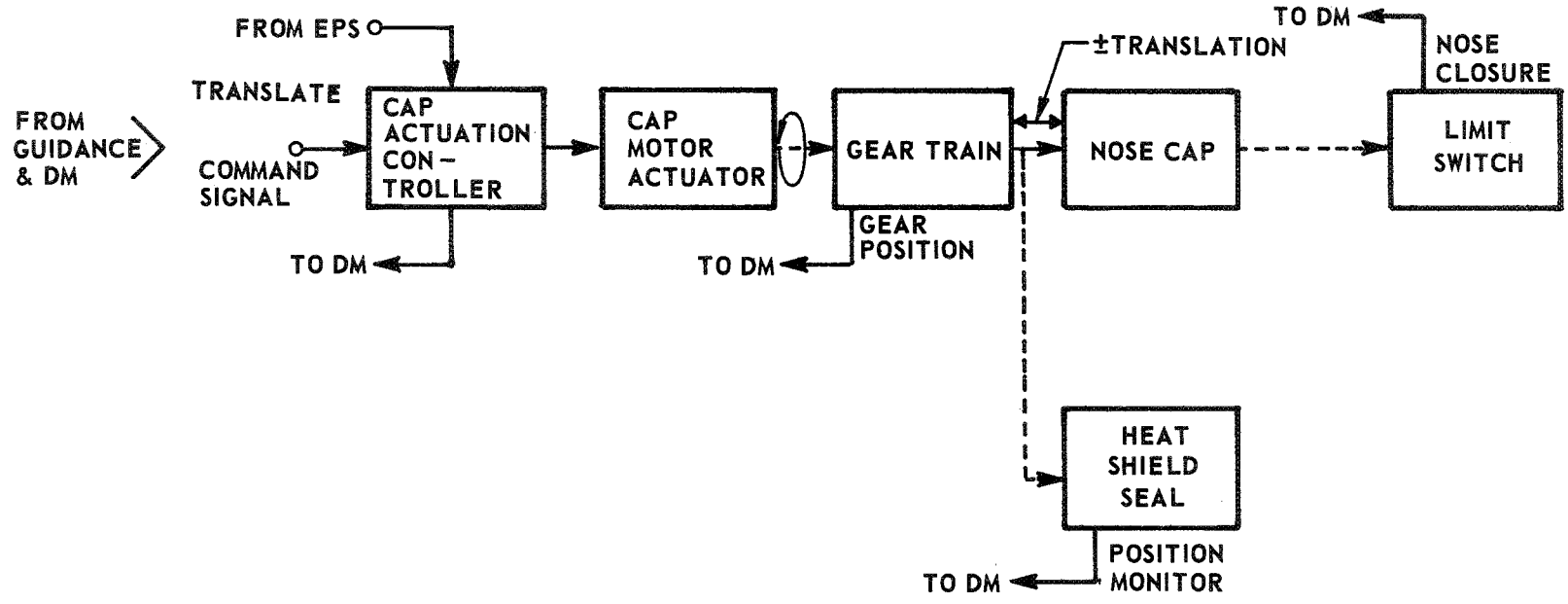


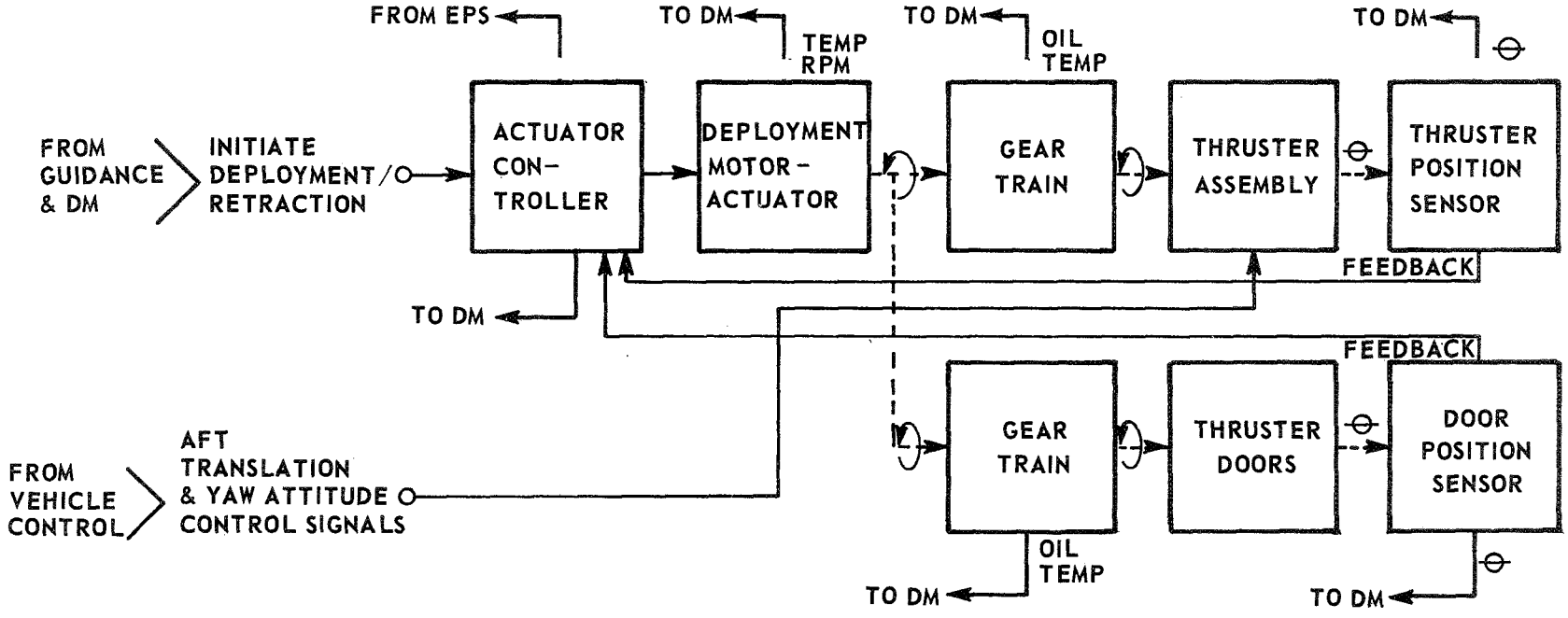
Figure D-1. 2. Vehicle separation control.



NOTES:

1. DM DATA MANAGEMENT
2. EPS= ELECTRICAL POWER SYSTEM

Figure D-1. 3. Nose cap control.



NOTES:

1. DM DATA MANAGEMENT
2. EPS ELECTRICAL POWER SYSTEM
3. ⊖ = DEPLOYMENT ANGLE

Figure D-1.4. Thruster deployment control.

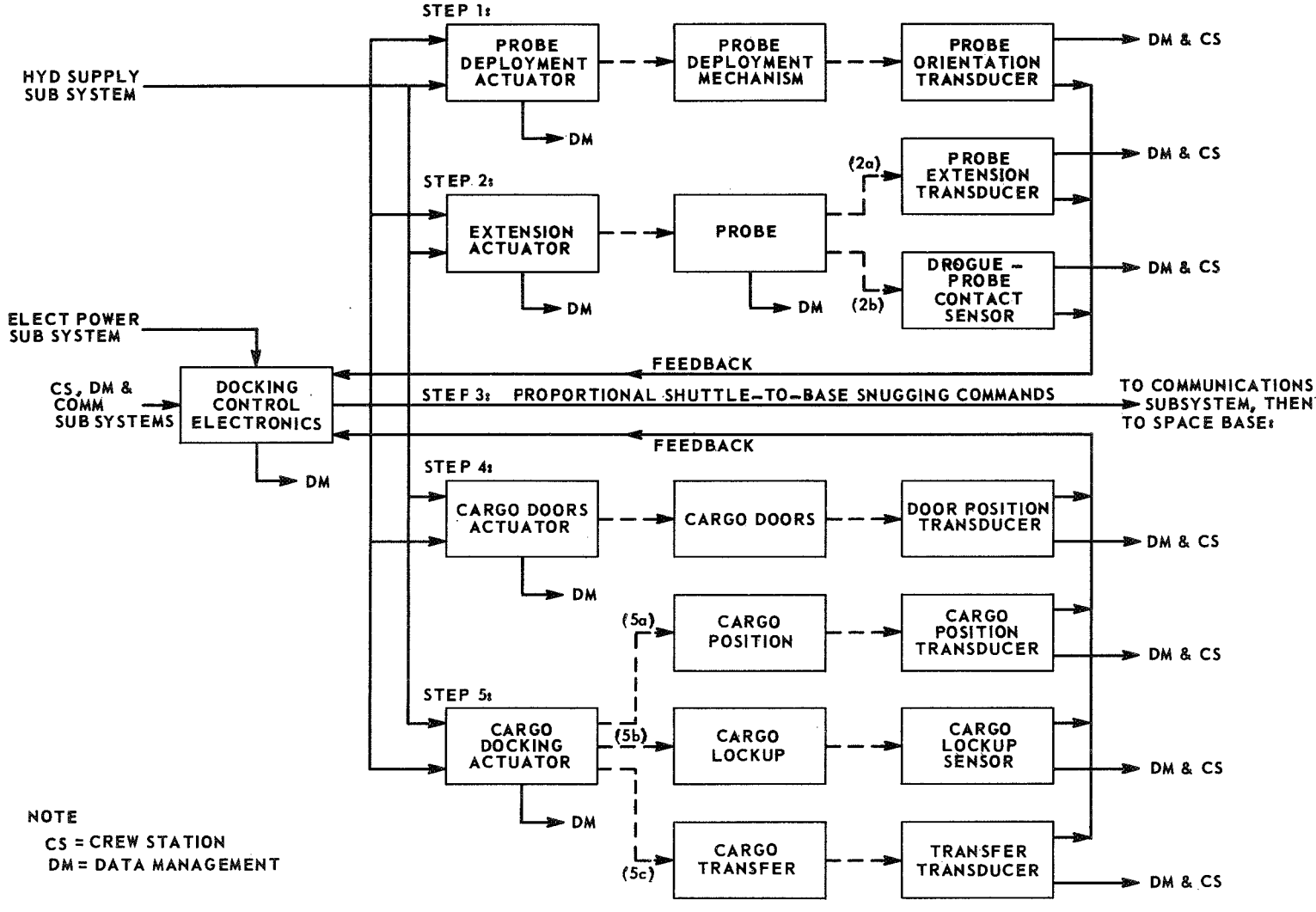


Figure D-1. 5. Docking adapter control.

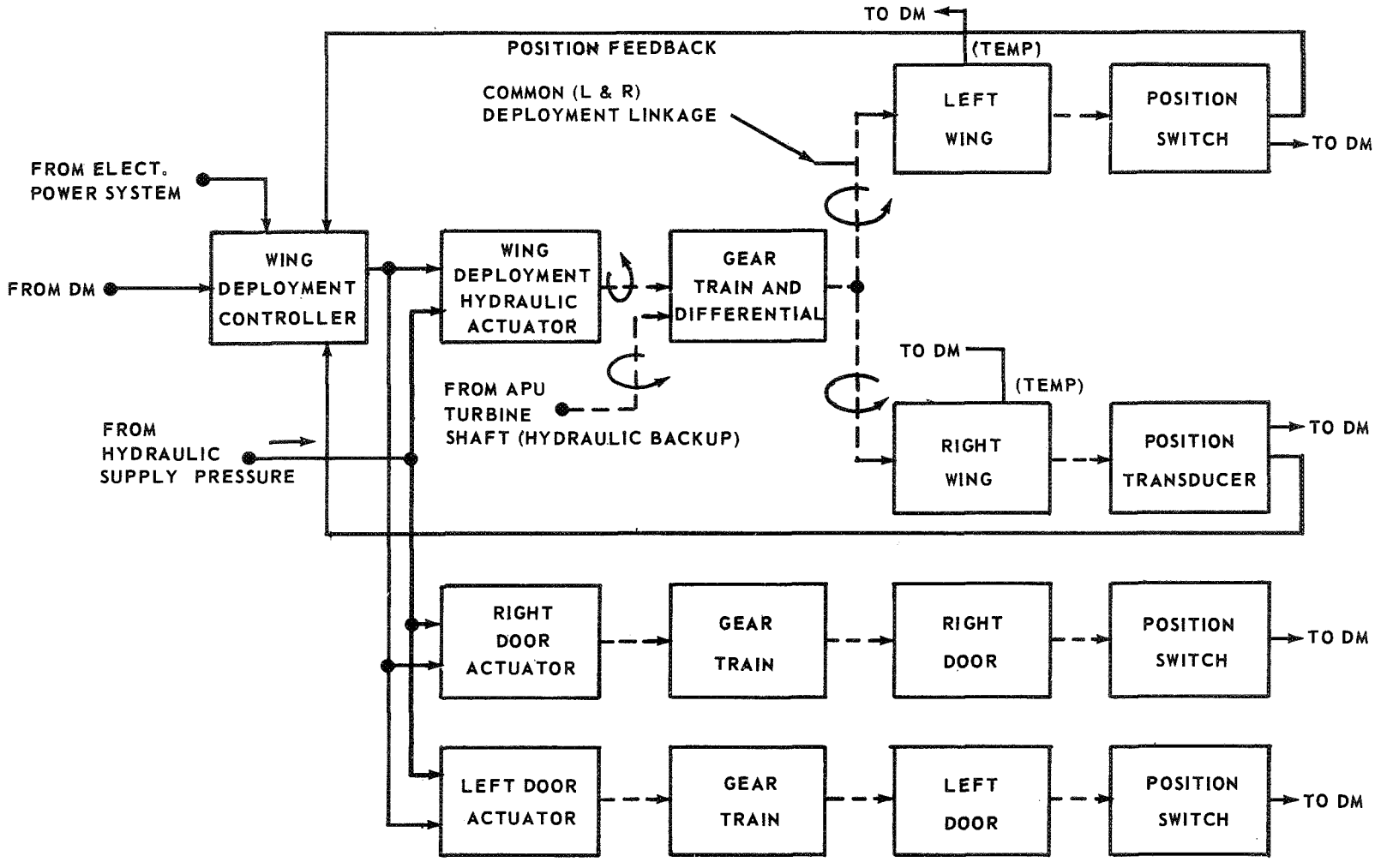
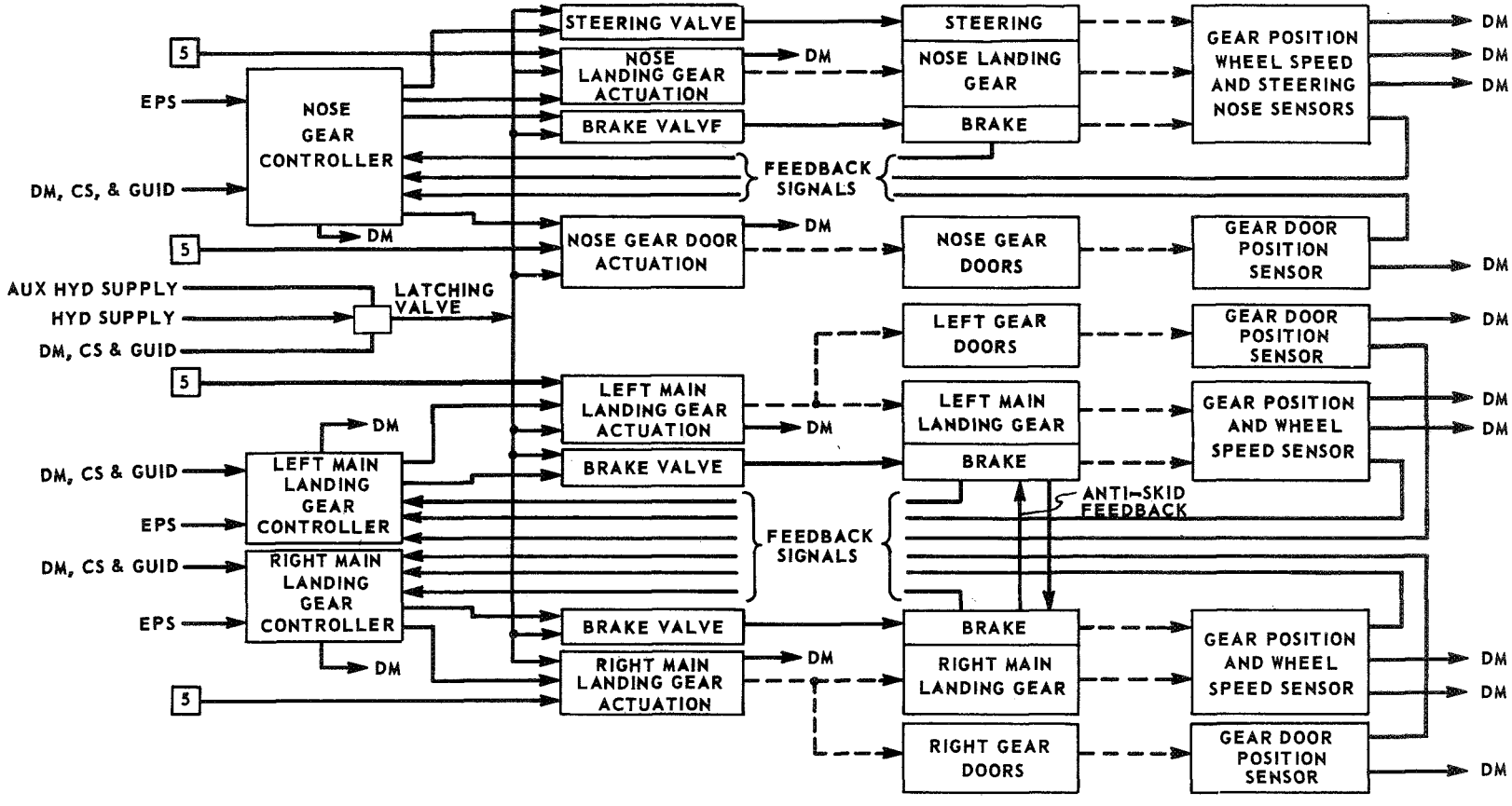
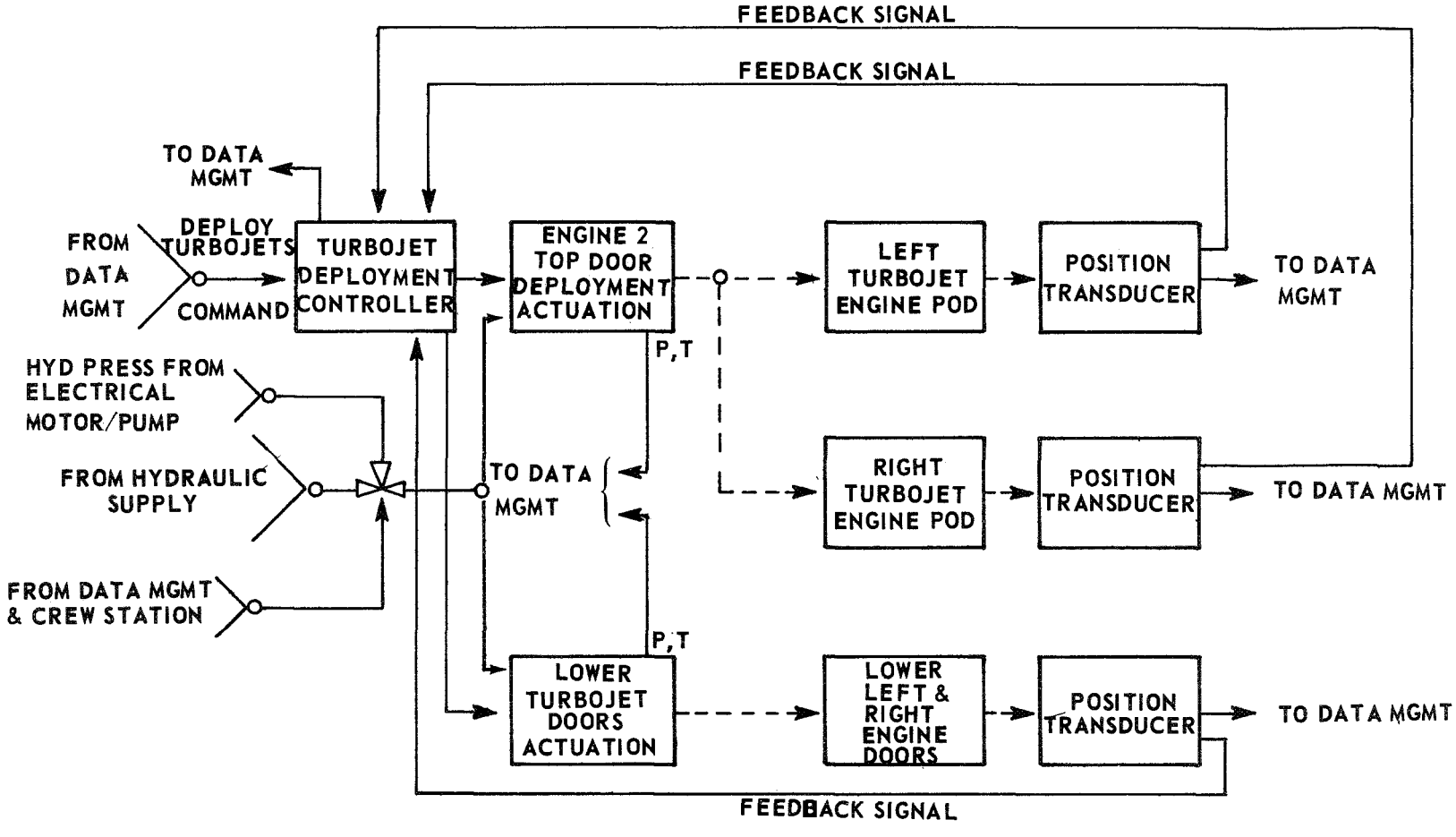


Figure D-1. 6. Wing deployment.



NOTES:
1. DM – DATA MANAGEMENT
2. CS – CREW SYSTEM
3. EPS – ELECTRICAL POWER SYSTEM
4. TYPICAL SIGNALS MIGHT BE, OPEN DOORS, DEPLOY GEAR, POSITION BOGIE, LOCK GEAR, APPLY BRAKING, ETC.
5. POSSIBLE BACKUP (OR PARALLEL REDUNDANT) CONTROL SIGNALS FROM CREW STATION (HYDRAULIC)

Figure D-1.7. Landing gear control.



NOTES:

APU = AUXILIARY POWER UNIT

Figure D-1. 8. Landing engines deployment.

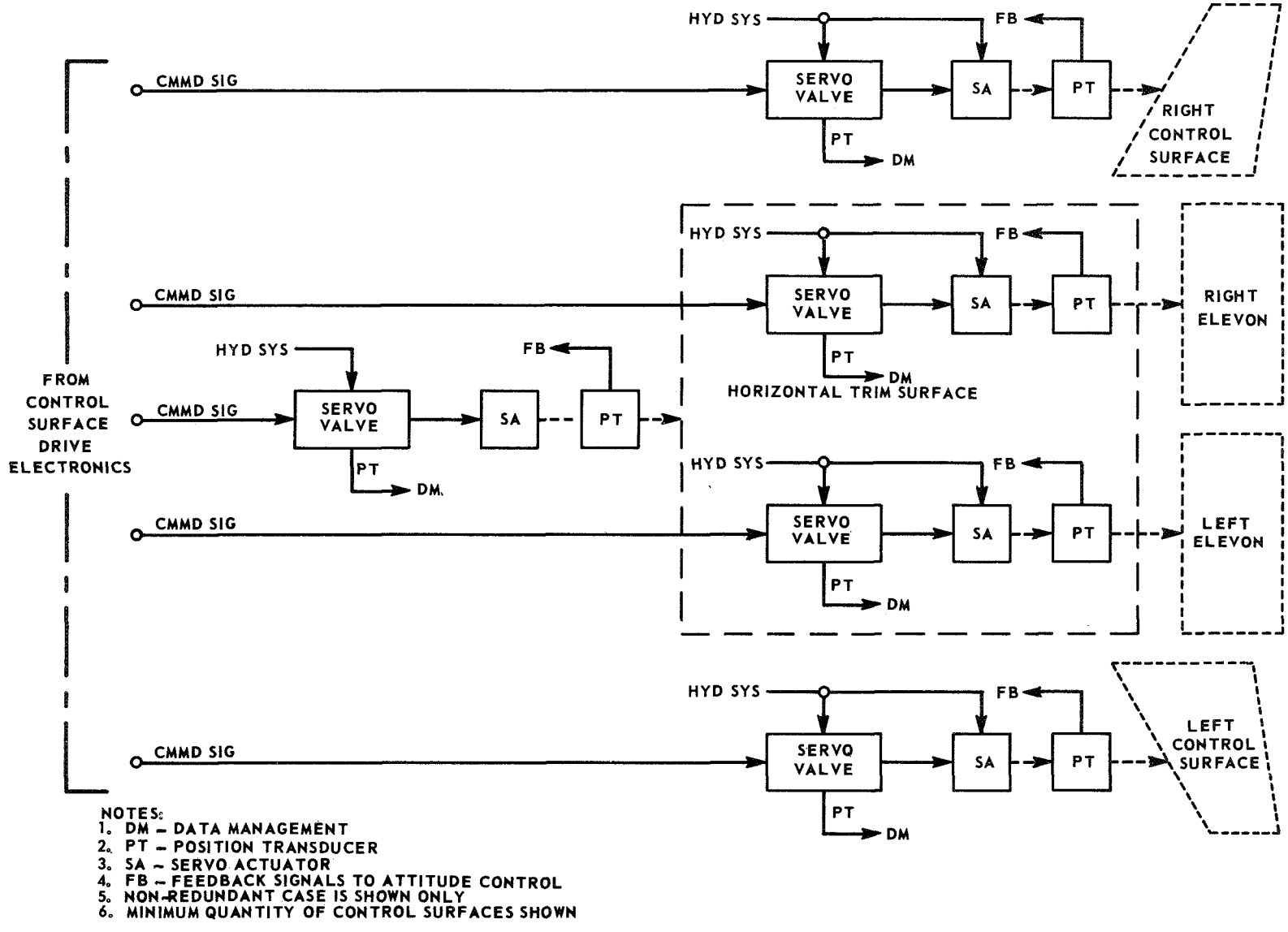


Figure D-1.9. Vehicle control surfaces.

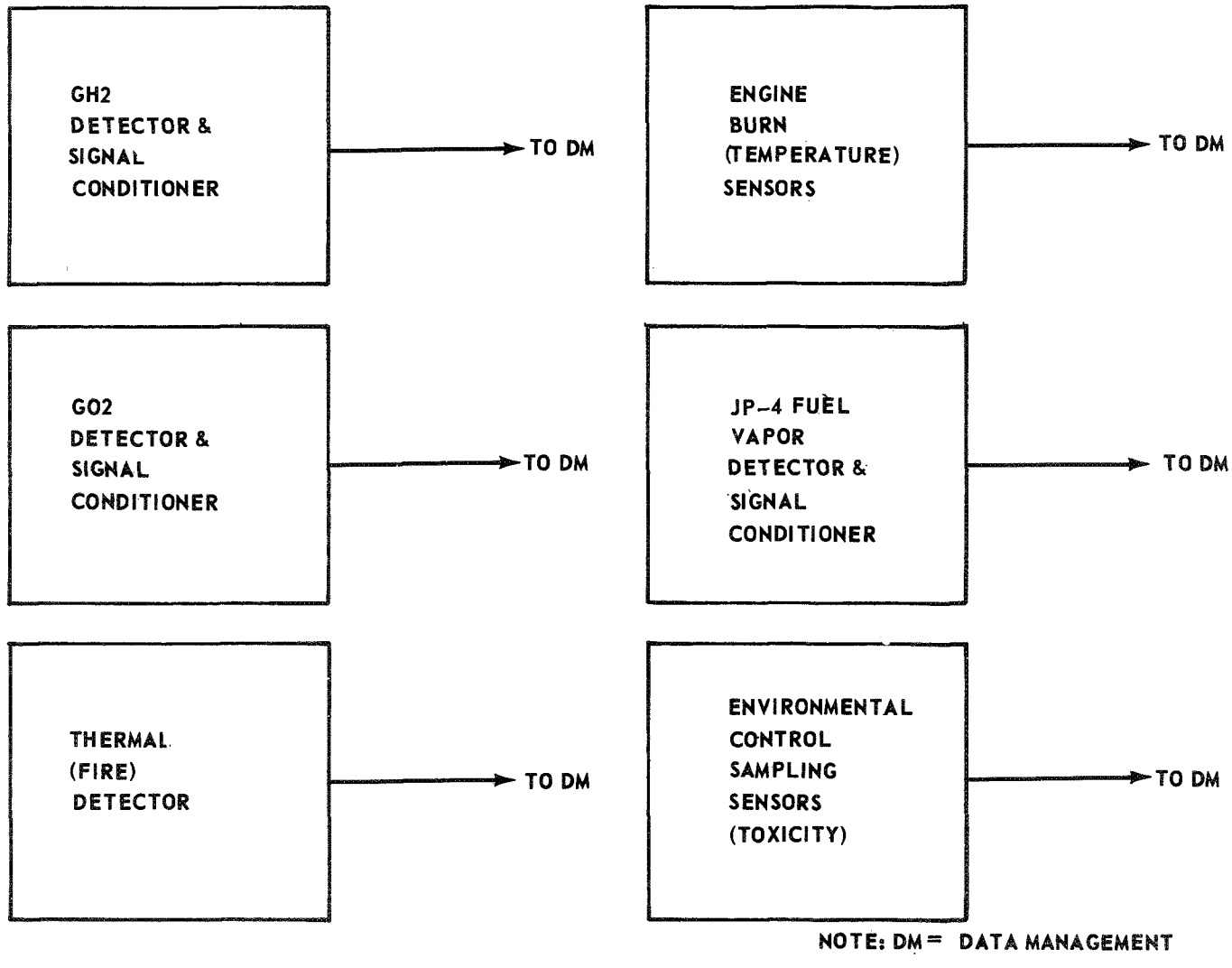


Figure D-1.10. Vehicle abort sensors.

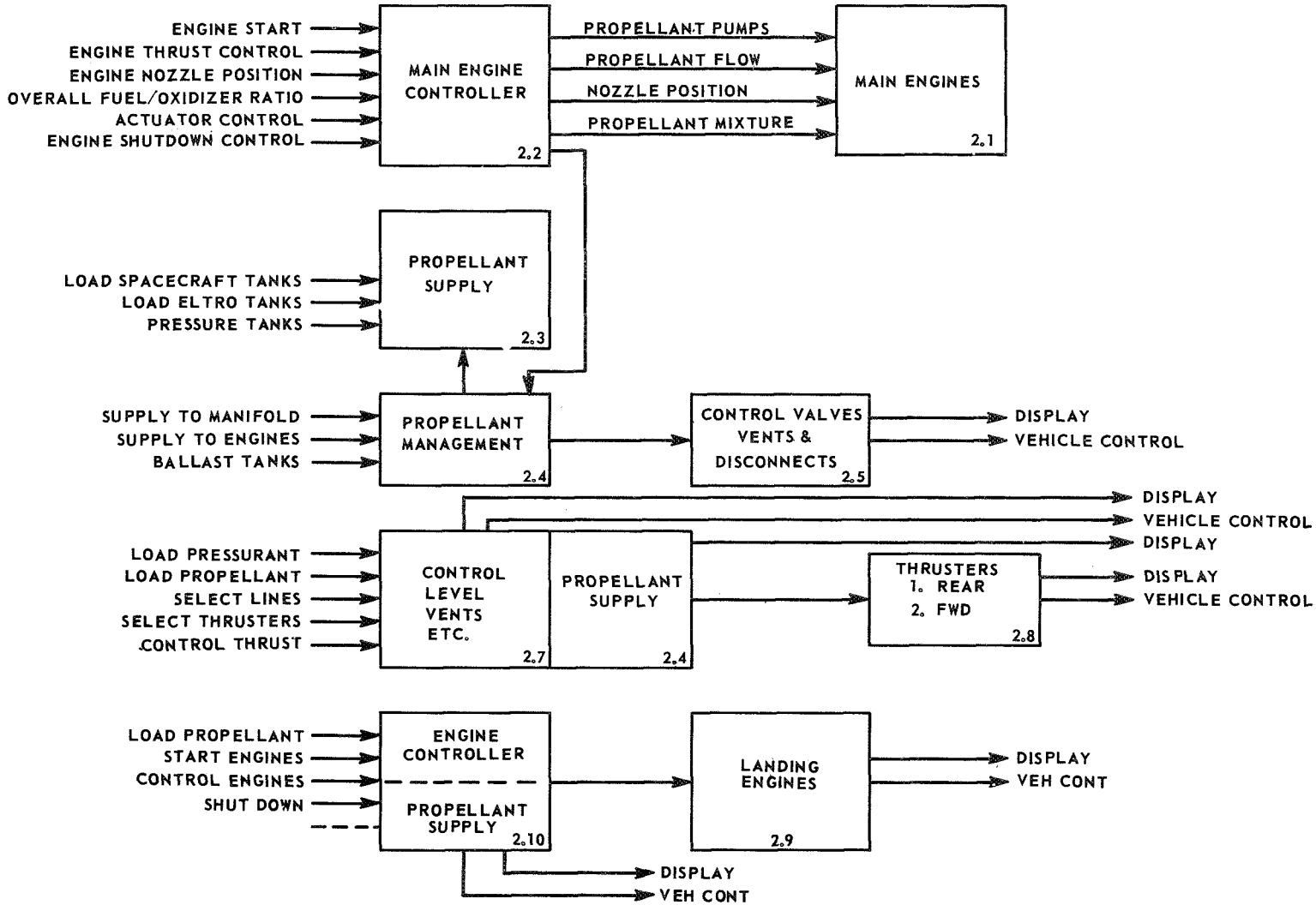


Figure D-2.0. Propulsion.

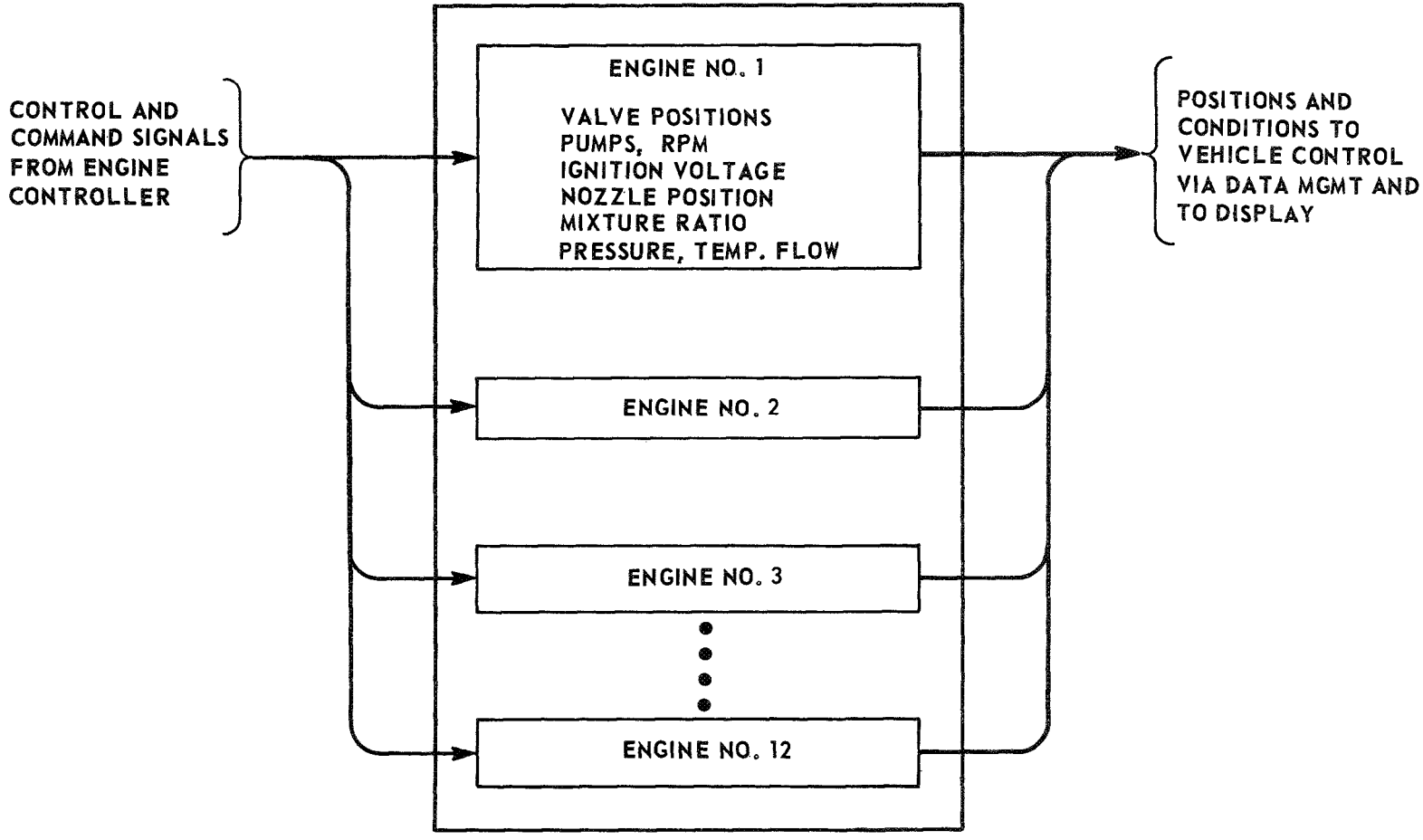


Figure D-2. 1. Main engines (12).

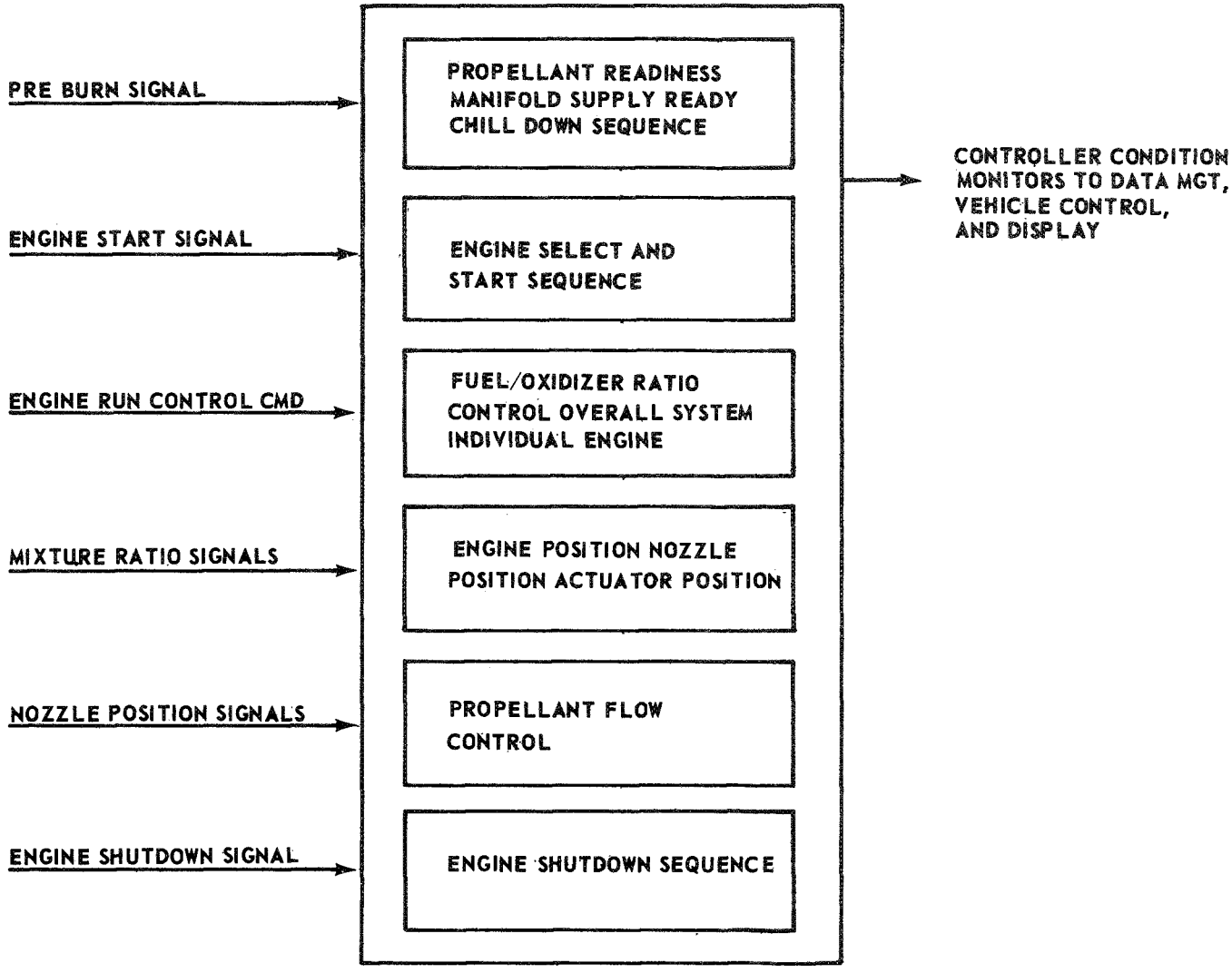


Figure D-2. 2. Main engine controller.

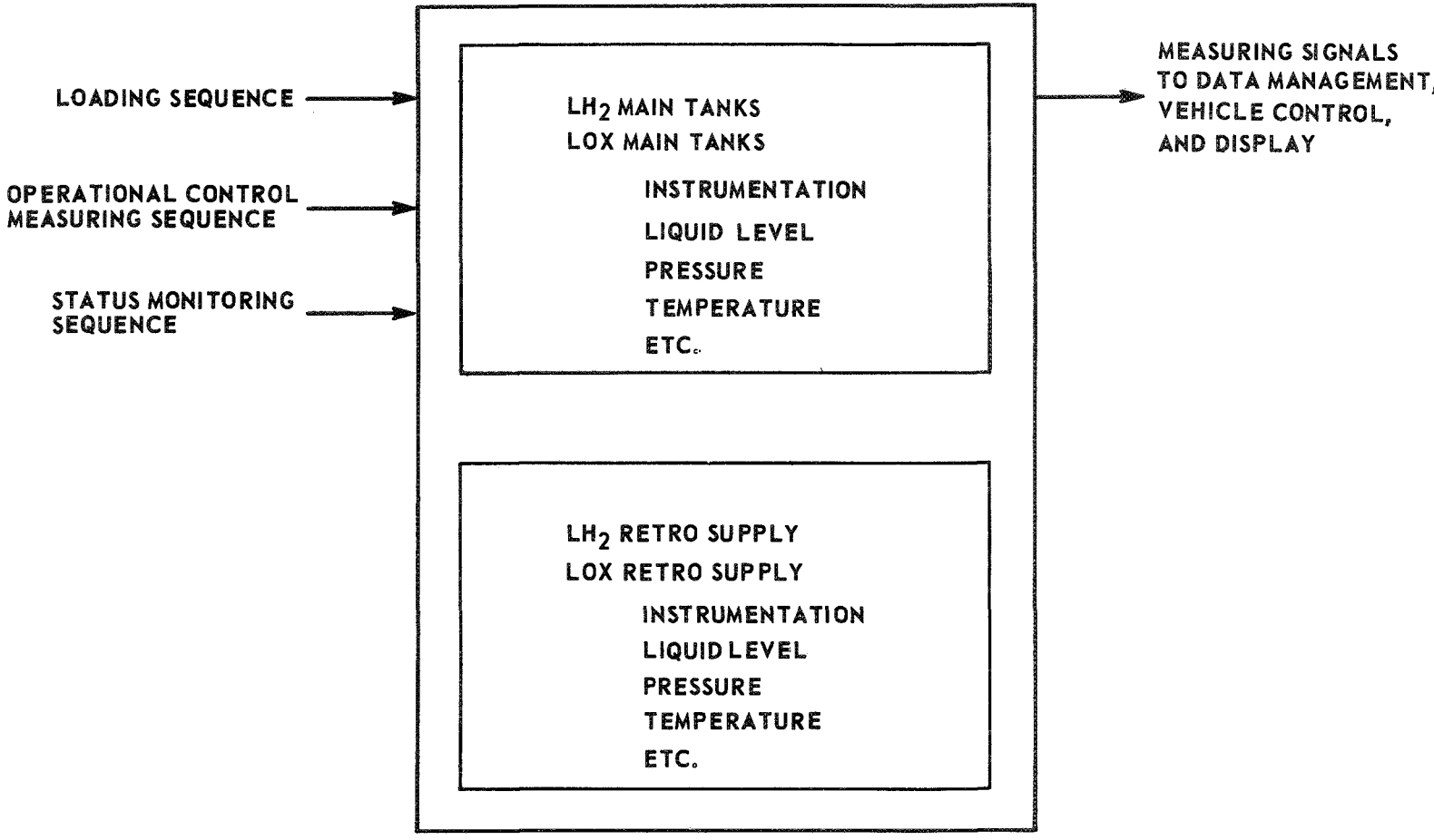


Figure D-2. 3. Propellant supply.

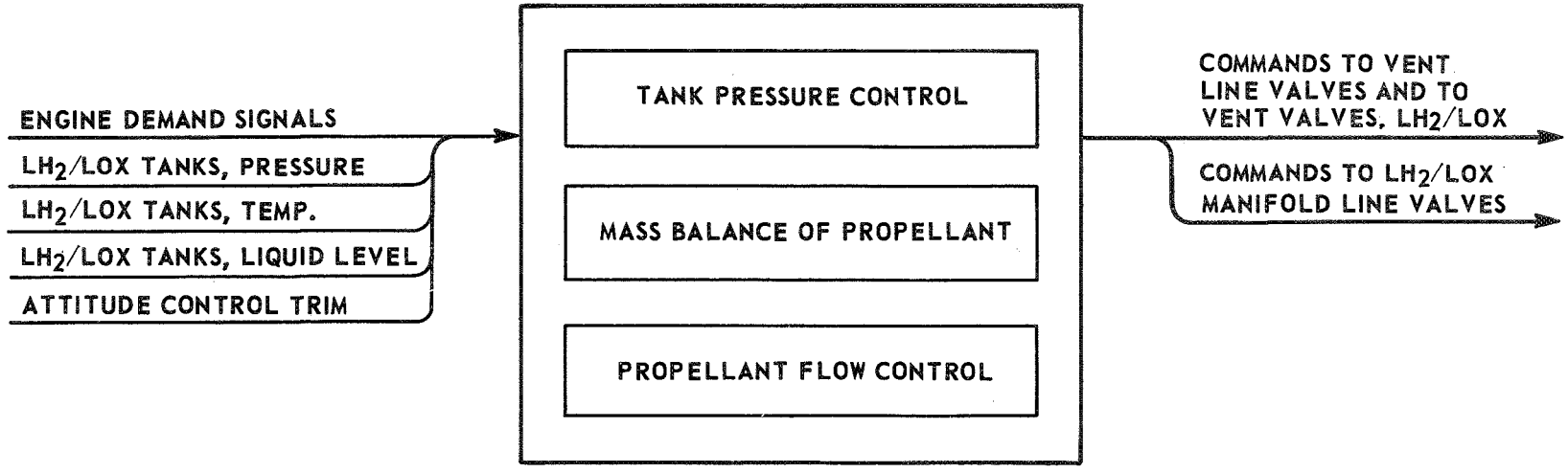


Figure D-2. 4. Propellant management.

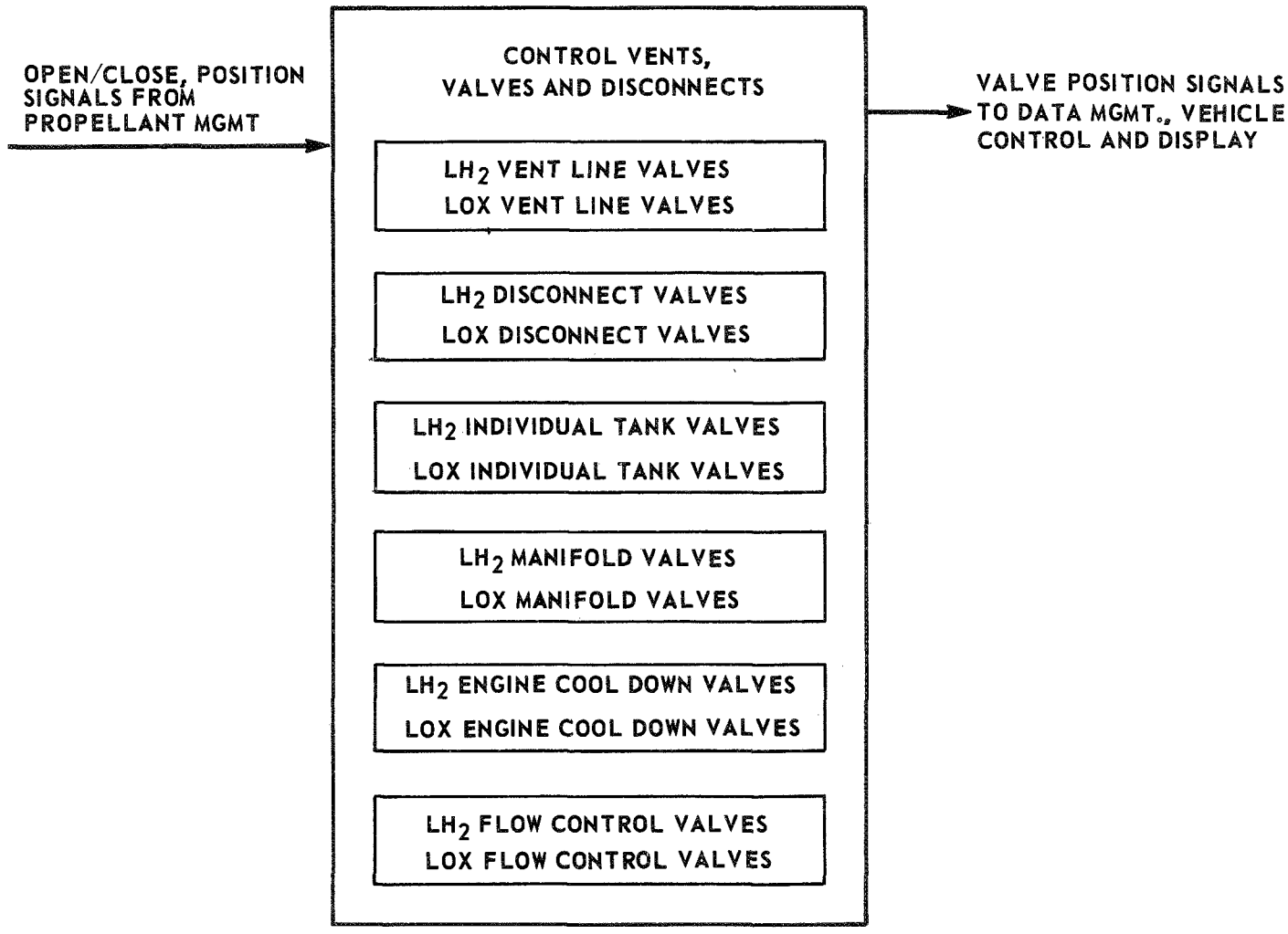


Figure D-2. 5. Propellant control valving.

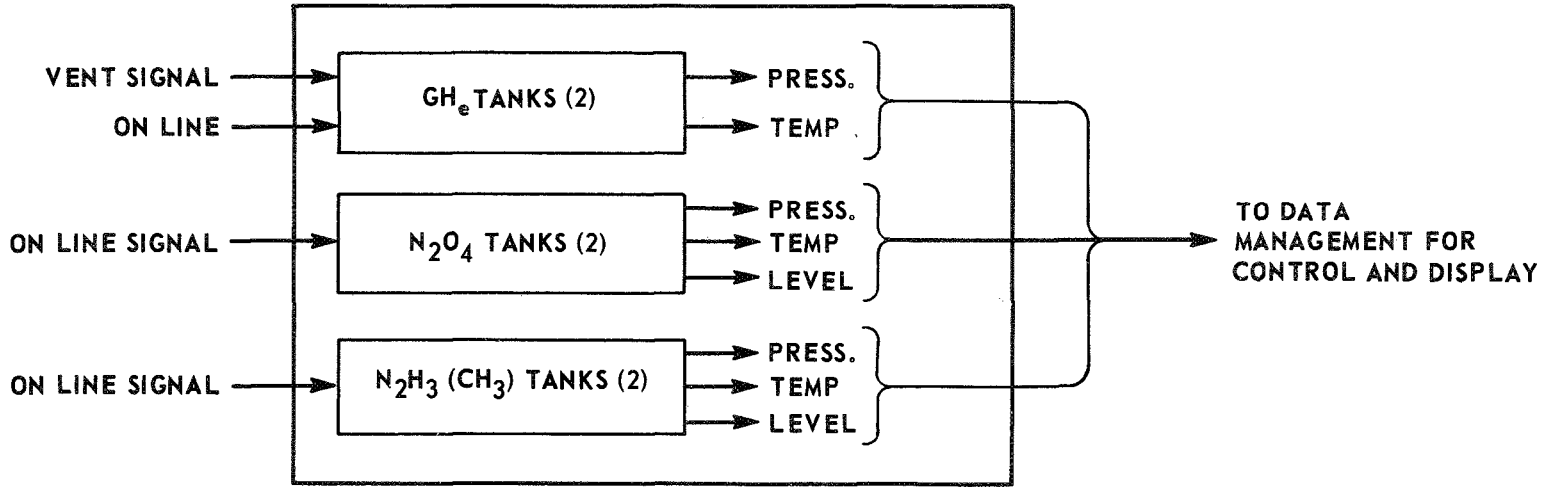


Figure D-2. 6. Propellant supply.

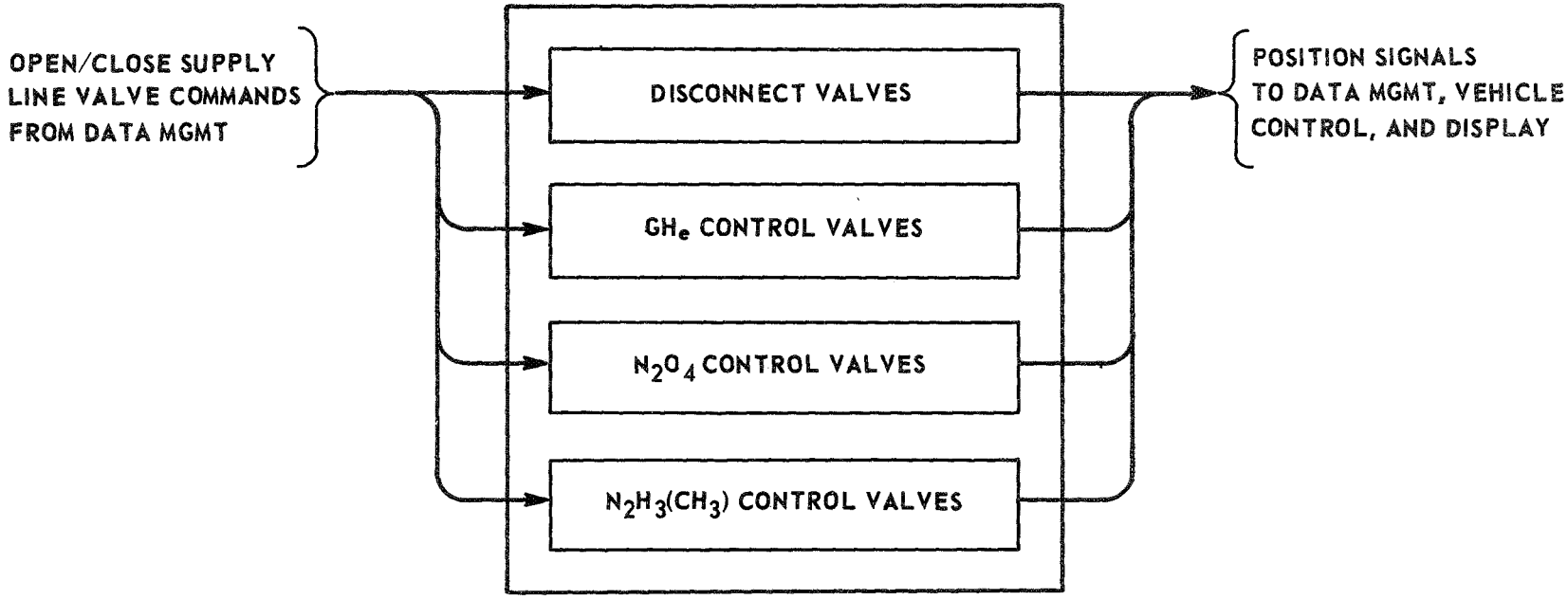


Figure D-2.7. Control level vents.

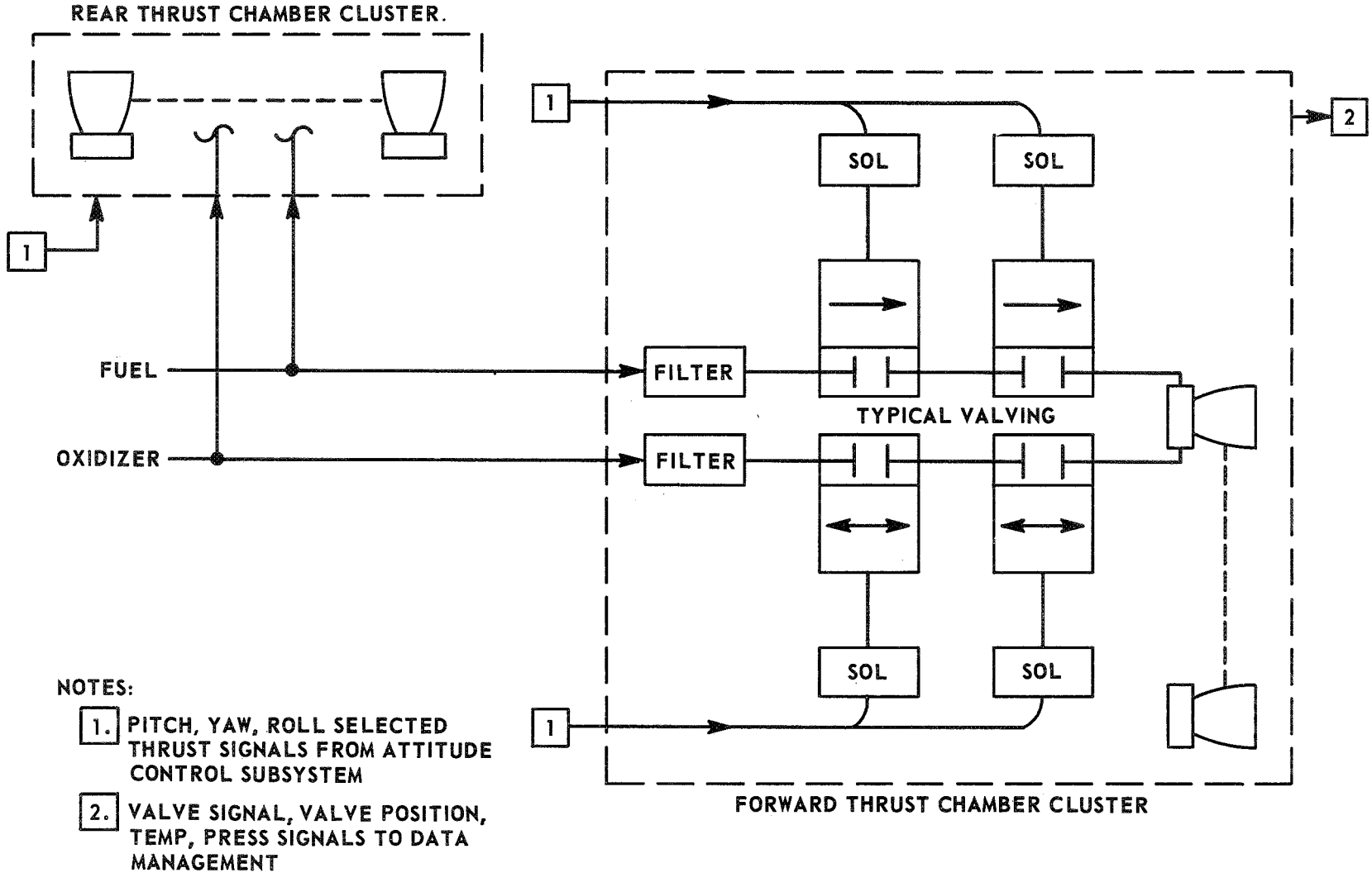


Figure D-2. 8. Thrusters.

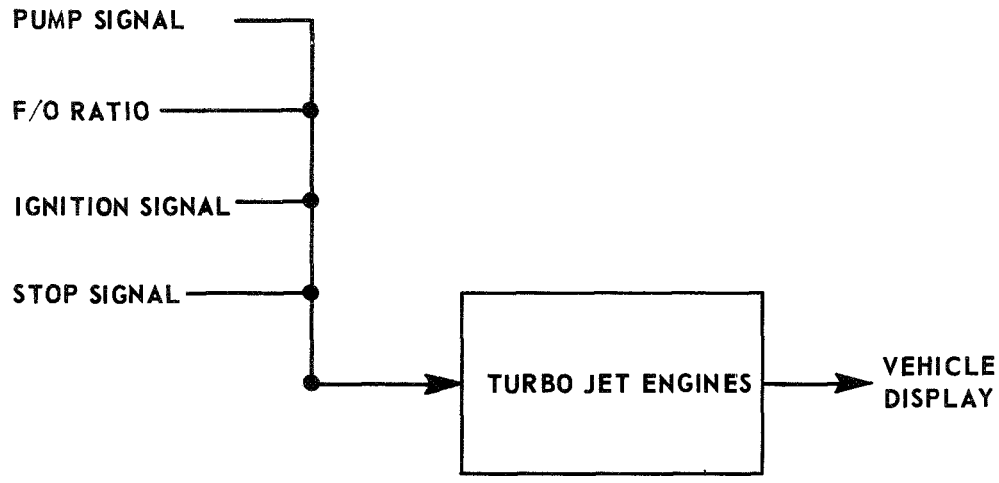


Figure D-2. 9. Landing-aid engines.

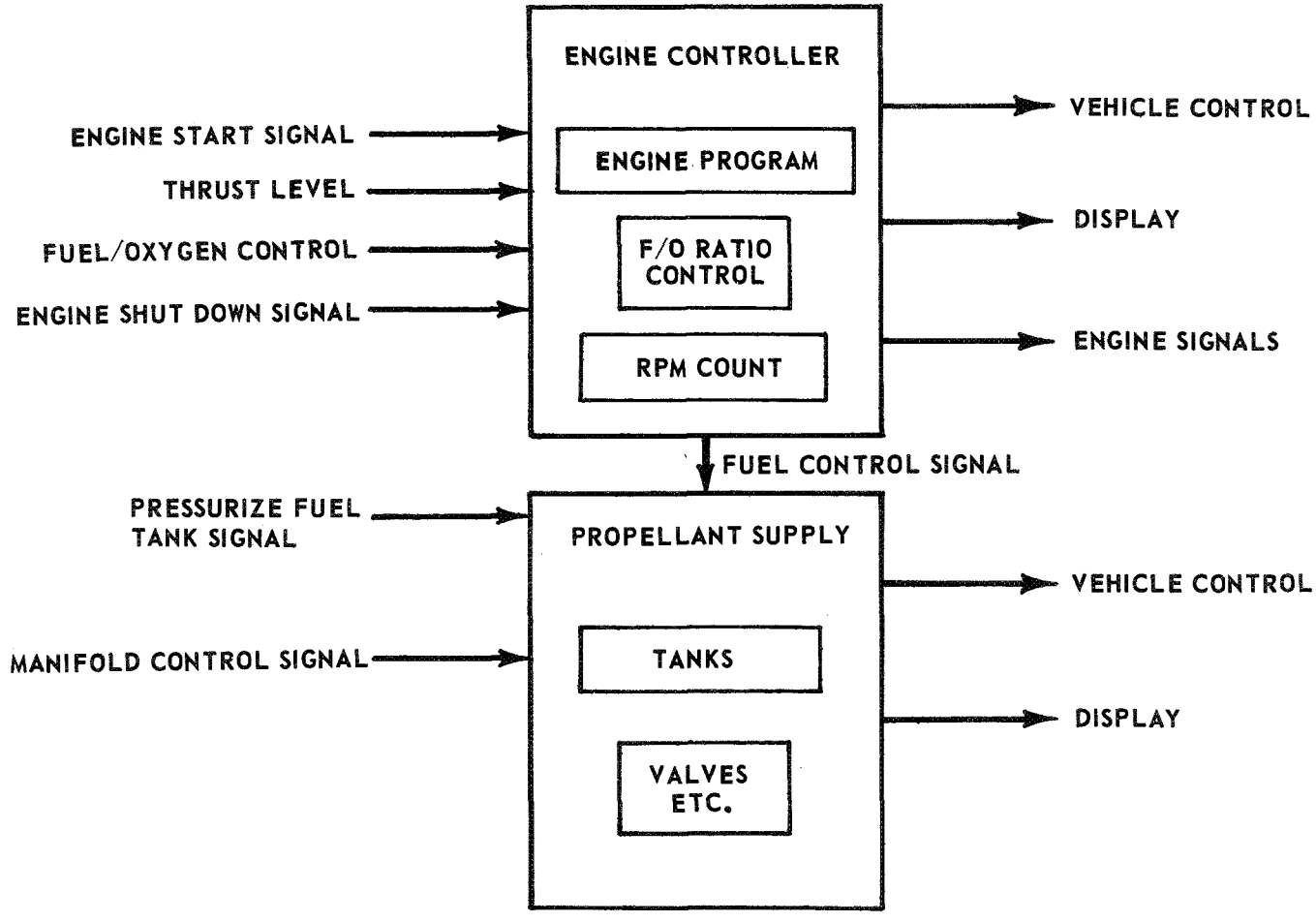


Figure D-2. 10. Propellant supply/controller.

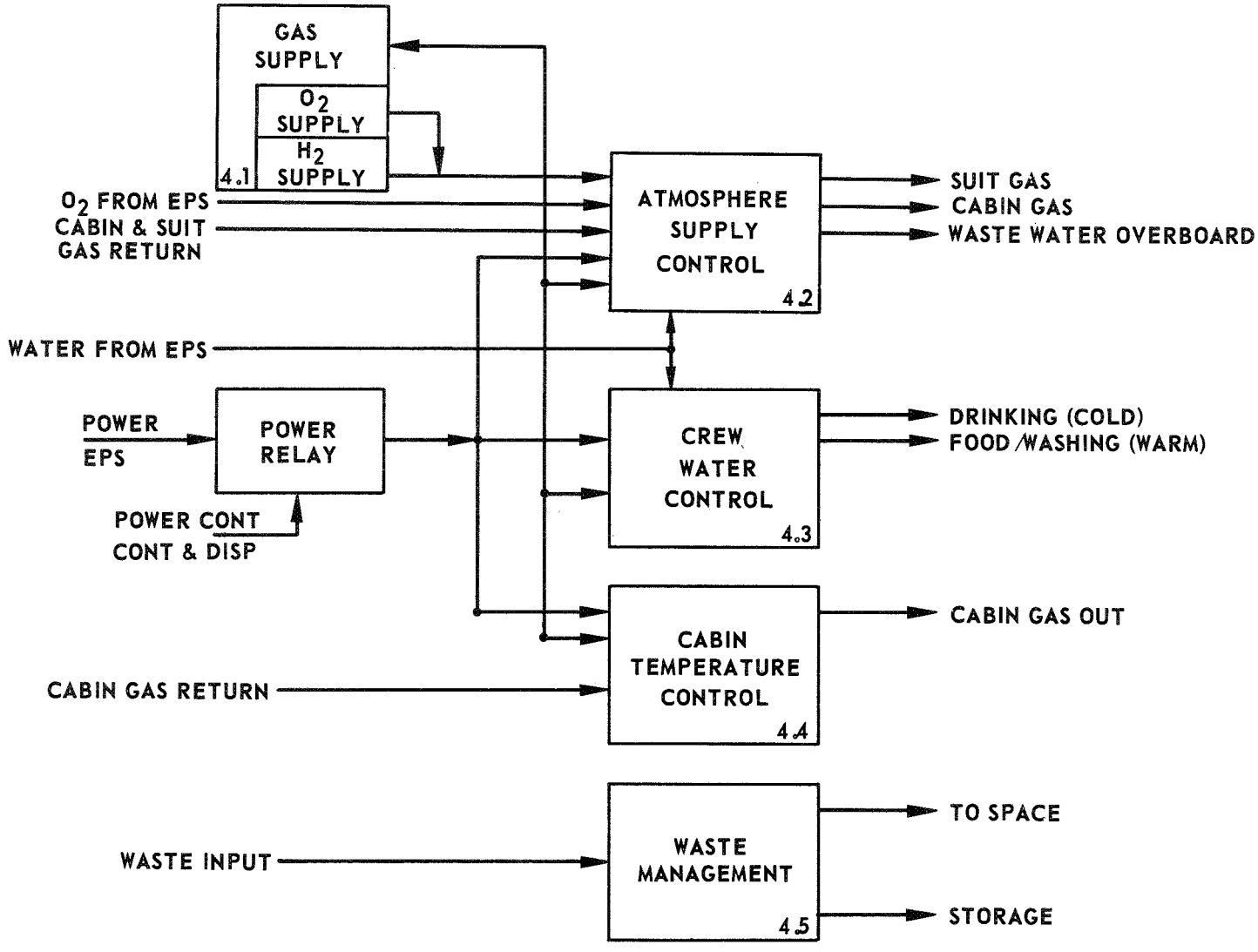


Figure D-4.0. Environmental control system.

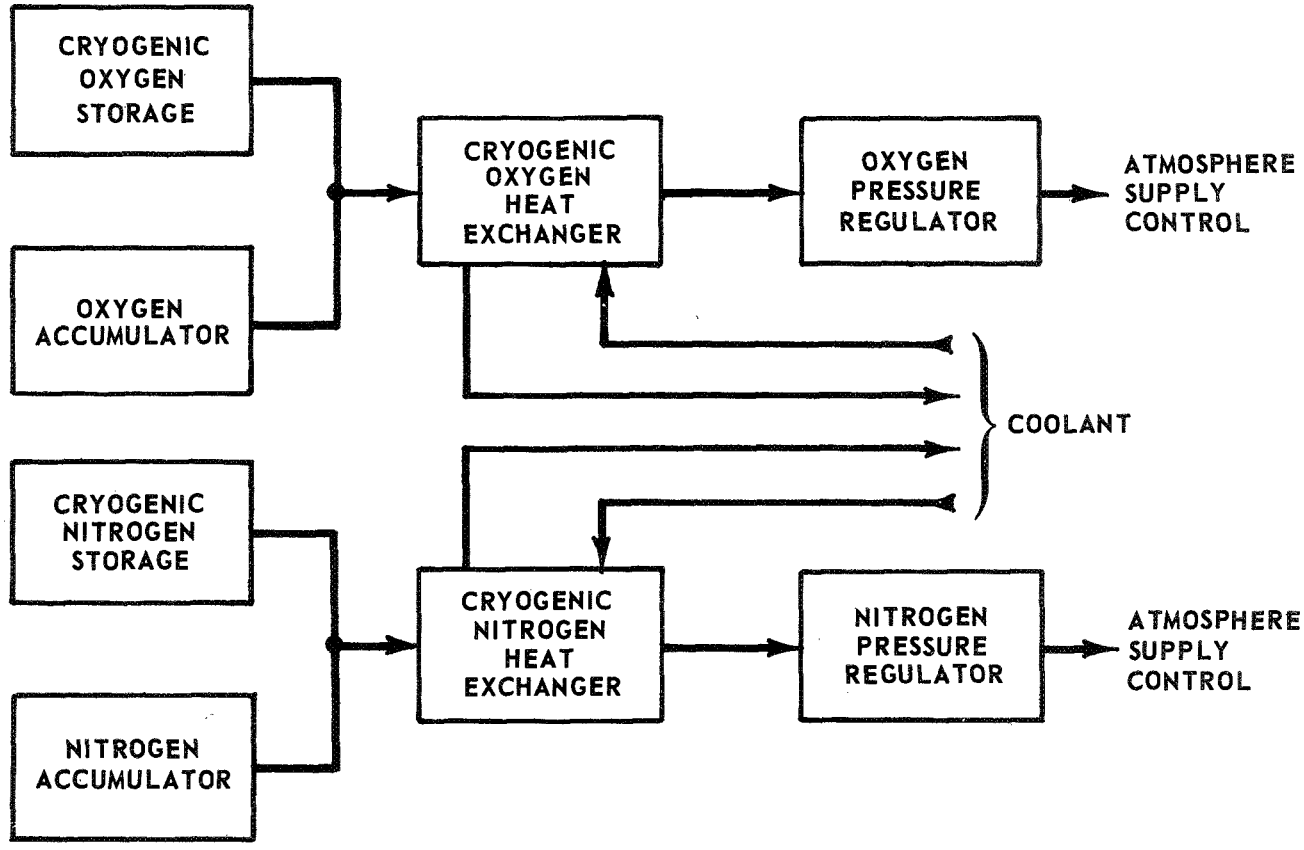


Figure D-4.1. Gas supply.

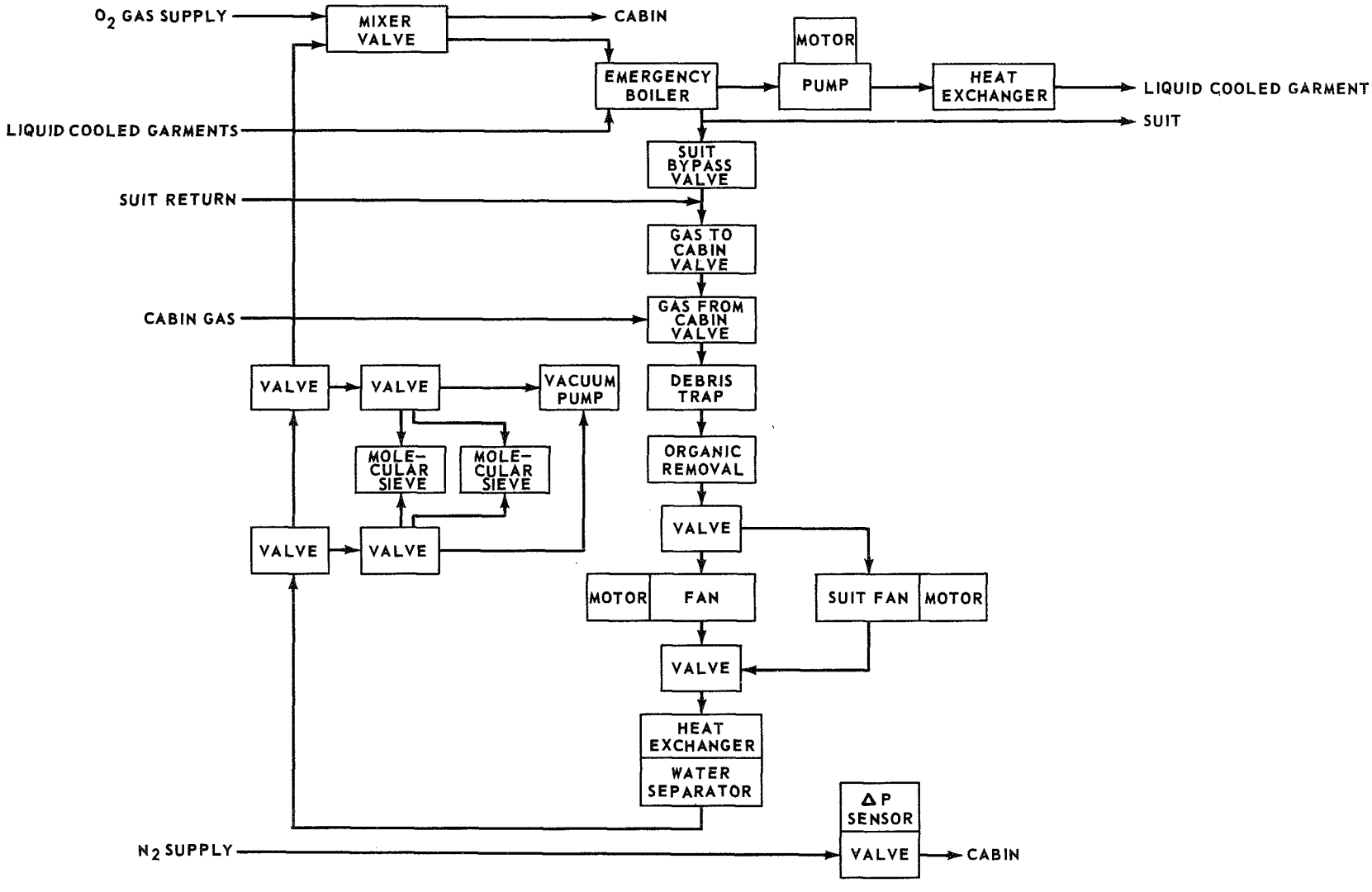


Figure D-4. 2. Atmosphere supply control.

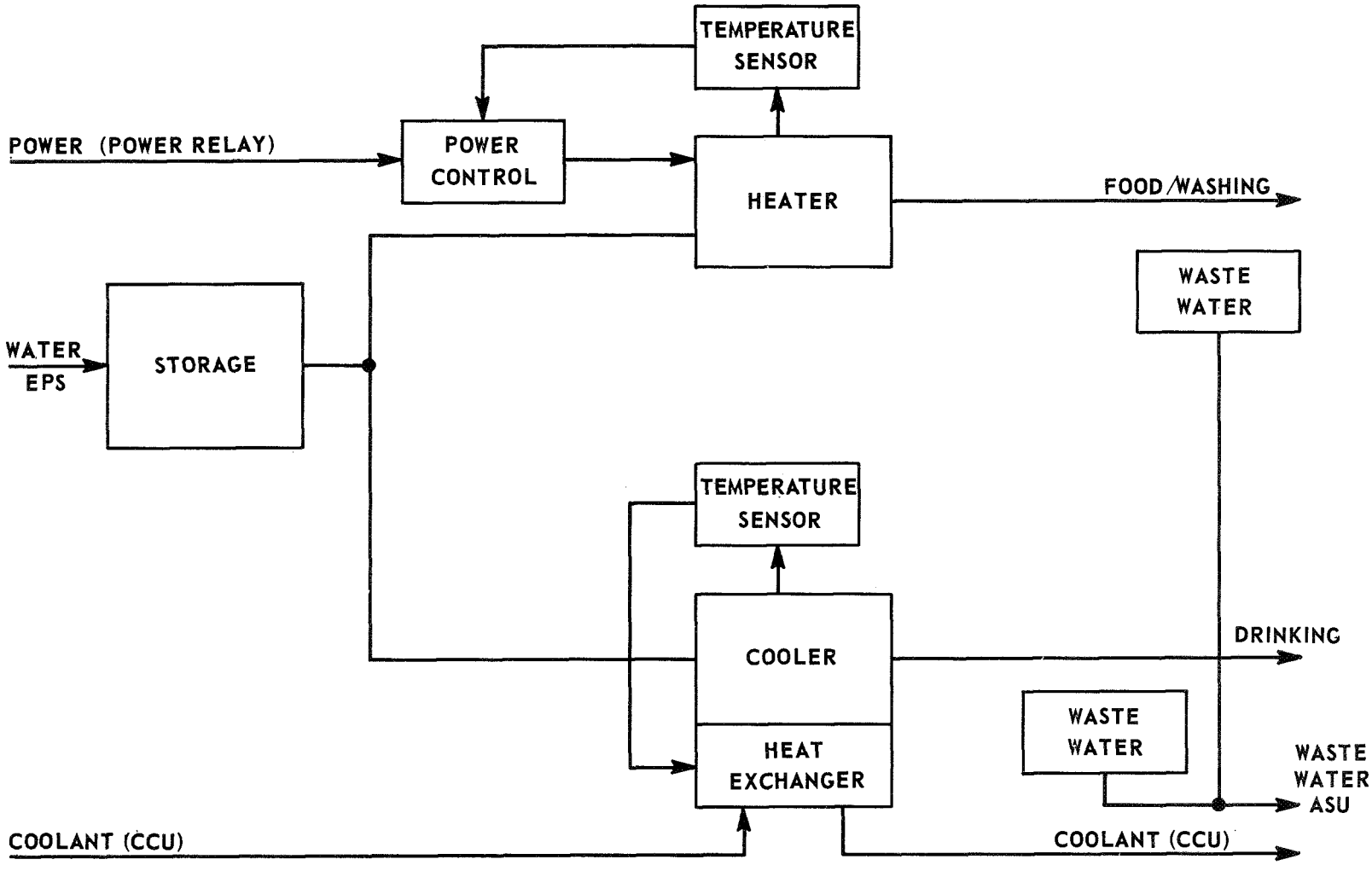


Figure D-4. 3. Crew water control.

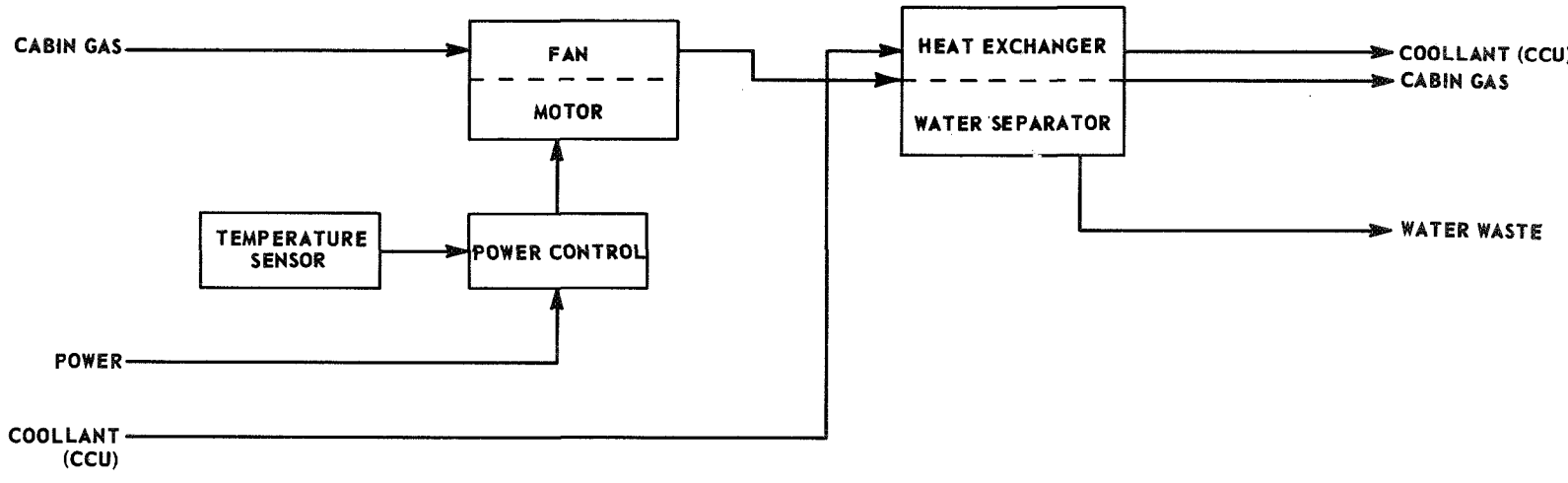


Figure D-4. 4. Cabin temperature control.

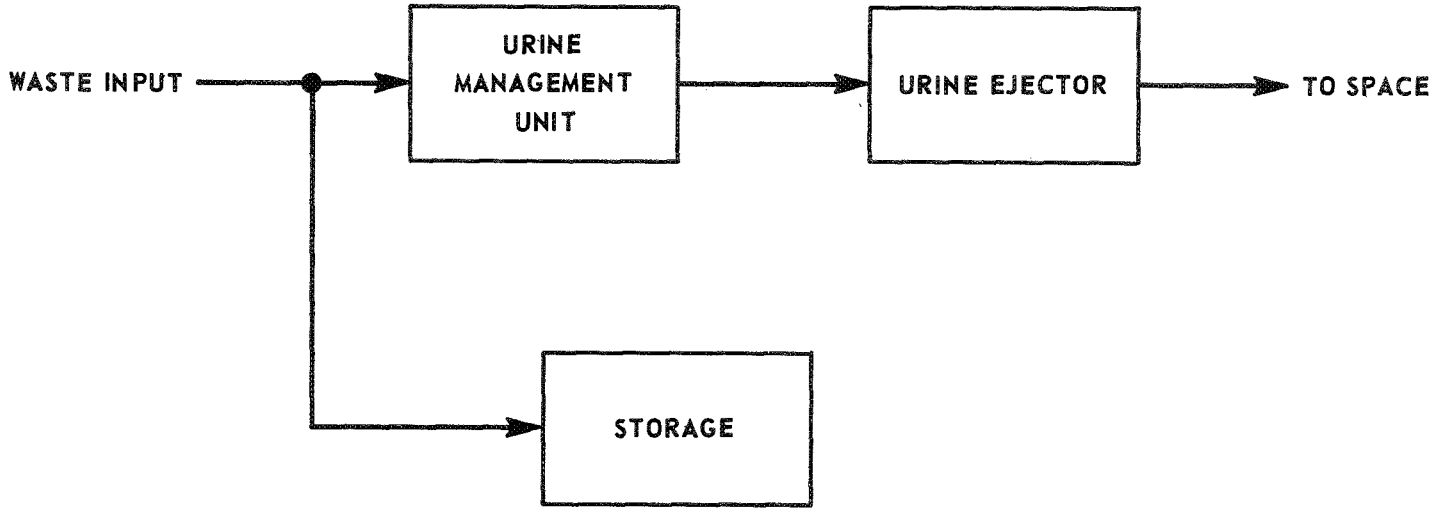


Figure D-4. 5. Waste management.

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APPROVAL

A CONCEPTUAL DESIGN OF THE SPACE SHUTTLE INTEGRATED AVIONICS SYSTEM

By Fred E. Digesu (editor)

The information in this report has been reviewed for security classification. Review of any information concerning Department of Defense or Atomic Energy Commission programs has been made by the MSFC Security Classification Officer. This report, in its entirety, has been determined to be unclassified.

This document has also been reviewed and approved for technical accuracy.

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